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Rotorcraft Design for Operations

NORTH ATLANTIC TREATY ORGANIZATION

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.423

ROTORCRAFT DESIGN FOR OPERATIONS

Papers presented at the Flight Mechanics Panel Symposium held in Amsterdam, Netherlands, from 13 to 16 October 1986.

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PREFACE

The expanding roles of the helicopter and the intensified threat perceived by its potential users have led to proposals for future rotorcraft with characteristics significantly different to existing types. The resulting rapid evolution of rotorcraft configurations, in response to user demands, now requires a translation into design criteria to permit the aerospace R & D community to provide appropriate and cost effective responses to these demands. The objective of this symposium was to explore the impact of operational needs on the evolution of rotorcraft design and to identify priorities and neglected topics. Three specific issues were central:

— The translation of operational mission requirements into design criteria

— The evaluation of techniques to incorporate user defined needs into the design and methods of test and verification

— The identification of design areas where unusual or new user needs are demanding special or radical features.

All papers were obtained by invitation.

This conference and these proceedings were assessed in a Technical Evaluation Report, commissioned by the AGARD Flight Mechanics Panel and published separately as an Executive Summary as AGARD AR-243.

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Le rôle croissant de l'hélicoptère et de la menace, de plus en plus pressante, perçue par les utilisateurs potentiels ont conduit à la formulation de propositions pour des futures voilures tournantes dont les caractéristiques seront très différentes de celles des modèles existants. En réponse aux demandes de l'utilisateur l'évolution rapide des configurations des voilures tournantes impose maintenant la traduction en critères d'étude, afin de permettre à la communauté de R et D aéronautique de fournir les réponses appropriées et le coût réel de ces exigences. L'objectif de ce symposium avait pour objet de passer en revue l'état actuel de la conception des voilure tournantes et de mettre en exergue les priorités et les omissions. Trois points spécifiques étaient au centre du problème:

— la traduction des exigences requises pour les missions opérationnelles en critères conceptuels

— l'évaluation des techniques pour tenir compte des besoins de l'utilisateur au niveau de la conception et des méthodes d'essais et de vérification

— l'identification des domaines d'étude où les besoins nouveaux ou inhabituels de l'utilisateur demandent des caractéristiques spéciales ou essentiellement différentes.

Toutes les communications ont été obtenues par voie d'invitation.

Le compte rendu du symposium demandé par la commission Mécanique du Vol de l'AGARD contient un rapport d'évaluation technique qui est aussi disponible de facon séparée sous forme d'un résumé entitulé "Executive Summary. AGARD AR-243".

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LE DIALOGUE "OPERATIONNELS-INGENIEURS"

par

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Je voudrais d'abord vous dire combien je suis honoré et heureux de participer à ce symposium et je remercie les organisateurs de me permettre de développer devant une aussi docte assemblée un sujet qui me tient à coeur depuis de nombreuses années, et qui s'inscrit dans le domaine beaucoup plus large des conditions à réaliser pour qu'un programme de matériels nouveaux soit mené à terme, à la satisfaction générale, je veux parler des relations qui doivent s'établir entre "tacticiens" et "techniciens" et plus précisément du

dialogue opérationnels - ingénieurs
ou états-majors et directions ou services techniques.

Il faut bien reconnaître que ce "dialogue" a toujours, et c'est encore le cas aujourd'hui, à un degré moindre cependant, fait l'objet de nombreuses critiques :

- on se plait souvent à souligner les préoccupations différentes de l'Etat-Major et des directions ou services techniques :

. Au premier, le souci d'obtenir un matériel répondant le mieux possible au besoin militaire,

. Aux seconds, la préoccupation d'abord d'assurer un plan de charge à certaines entreprises ou de vendre à l'extérieur, ce qui conduit parfois à "négliger" en partie la satisfaction du besoin militaire.

A un autre niveau et dans le langage familier des officiers chargés des programmes de matériels, trois affirmations reviennent régulièrement :

- "On n'a pas ce qu'on veut",
- "Ca arrive trop tard",
- "C'est trop cher",

sans pousser plus avant l'analyse des différentes responsabilités d'un tel état de choses.

Ceci n'est plus admissible et le dialogue "opérationnels - ingénieurs" doit constamment être amélioré, car, de sa qualité, peut dépendre la réussite ou l'échec d'un programme, et ceci est particulièrement vrai pour les hélicoptères.

Les données de base du dialogue opérationnels - ingénieurs ont été en effet profondément modifiées dans une période récente en donnant à ce dialogue d'ailleurs une dimension tout à fait nouvelle.

Je vous propose donc d'aborder successivement les points suivants:

- les nouvelles données du dialogue "opérationnels - ingénieurs",
- les responsabilités des opérationnels dans la définition des besoins et la conduite d'un programme,
- les responsabilités des ingénieurs dans la réalisation des outils techniques,
- les améliorations possibles du dialogue.

Au passage, je vous donnerai quelques précisions sur les concepts généraux actuels de l'ALAT Française dans le domaine de la réalisation des matériels futurs.

1) Les données nouvelles du dialogue opérationnels - ingénieurs

Trois facteurs importants nouveaux viennent de modifier profondément les données du dialogue opérationnels -ingénieurs tout en donnant à ce dialogue une autre dimension.

Le premier concerne le processus d'adaptation d'une réalisation technique à un besoin opérationnel.

Jusqu'à une période que l'on peut situer au milieu du dernier conflit mondial, c'est le besoin opérationnel qui provoquait ou accélérait un progrès technique dans un domaine ou une direction donnée : le besoin opérationnel précédait le progrès technique. Il ne fait aucun doute par exemple que c'est le besoin vital d'assurer la sécurité des Iles Britanniques qui est à l'origine des progrès rapides dans le domaine des radars.

Aujourd'hui on assiste tous les jours à une "explosion" de découvertes ou progrès techniques dans beaucoup de domaines ou de nombreuses directions. Il ne s'agit plus alors pour l'opérationnel de se contenter d'exprimer un besoin et de laisser à l'ingénieur le soin et la responsabilité de réaliser l'outil correspondant,mais il s'agit au contraire pour l'opérationnel et l'ingénieur ensemble de saisir, avant l'adversaire, le progrès technique qui aura les incidences les plus importantes dans le domaine opérationnel.

Le deuxième facteur concerne la réalisation de l'outil de combat lui-même. Jusqu'à nos jours, dans l'ALAT Française en tout cas, on a toujours adapté un armement à un hélicoptère existant, conçu d'ailleurs à des fins civiles :

on militarisait un hélicoptère civil (à une exception près : le SA 330 PUMA qui répondait lui à des caractéristiques militaires mais ce n'était pas un hélicoptère armé).

Il est évident aujourd'hui que l'on ne peut disposer d'un hélicoptère de combat performant qu'à condition de construire l'hélicoptère autour et en fonction du système d'armes retenu. Je ne donnerai qu'un seul exemple :

on ne construit pas le même hélicoptère AC suivant que l'on décide de l'équiper d'une optique de nez, de toit ou de mât.

La troisième donnée enfin, et j'ai gardé le meilleur pour la fin, c'est l'importance prise par le facteur coût qui, à lui seul, s'il n'est pas ou s'il est mal maîtrisé, peut remettre en cause la poursuite d'un programme.

11) Les responsabilités des "opérationnels :

Voyons donc à la lumière de ces trois données de base les responsabilités des "opérationnels"; elles s'exercent dans deux domaines importants

- la définition du Besoin Opérationnel,
- le déroulement lui-même du programme.

Il y a dans ces deux domaines des fautes majeures à ne pas commettre sous peine de compromettre un programme dans le plus mauvais des cas ou, au minimum, d'en augmenter les coûts dans d'importantes proportions. M'adressant à une assemblée essentiellement composée d'ingénieurs, donc de responsables de la réalisation de matériels à qui on reproche souvent de "fabriquer trop cher", je le dis nettement, et, croyez-moi, ce n'est pas du tout pour vous faire plaisir :

"l'opérationnel" a souvent aussi une large part de responsabilité dans les coûts excessifs des matériels, soit parce que le besoin a été mal apprécié ou insuffisamment précisé, soit parce que le programme lui-même a été mal conduit.

22) L'appréciation et l'expression du Besoin : du danger d'exigences exorbitantes.

On ne soulignera jamais assez l'importance de cette première expression du besoin que constitue la "fiche de caractéristiques militaires"(qui a surement son équivalent dans les autres armées).

C'est un document capital pour la rédaction duquel "l'opérationnel" doit faire preuve d'un grand réalisme en n'exigeant ni l'inutile, ni l'impossible ... tout en se projetant toujours dans l'avenir.

Combien de matériels étaient morts-nés dès le stade de l'expression du besoin car celui-ci avait été mal apprécié (au regard et du besoin opérationnel, et des possibilités techniques), et pour lesquels on a cependant dépensé des sommes importantes. Je donnerai dans ce domaine deux exemples :

- Le premier est celui du SA 330 PUMA dit "version complète" qui devait décoller et atterrir par visibilité nulle et plafond nul, sans aucune aide au sol, faire du vol en nuages et en atmosphère givrante, appareil isolé ou en formation ... (fiche de caractéristiques en date du 15.07.63). Celà a conduit à un effort de recherche très important et à la réalisation d'équipements prototypes inédits (radar détecteur d'obstacles SAIGA, système de "tenue de poste"...), mais l'appareil n'a pas vu le jour dans les délais prescrits.

- Le deuxième exemple est plus intéressant car il répondait à un besoin opérationnel important qui aurait pu être satisfait par l'équipement proposé si les performances demandées avaient été plus réalistes et si "l'opérationnel" avait eu une meilleure connaissance de l'évolution d'un hélicoptère dans le temps :
Je veux parler du "train automoteur DOP" étudié pour le SA 330 PUMA.(voir planche 1)
L'objectif était de permettre à l'hélicoptère de se déplacer au sol par ses propres moyens,sur de courtes distances bien sûr, pour rejoindre un couvert ou une zone d'ombre et bénéficier ainsi d'un bien meilleur camouflage. Un SA 330 (06) a été équipé d'un dispositif astucieux (à crémaillère et fonctionnant à partir de l'un des deux circuits hydrauliques de l'appareil) mais qui n'a pas été retenu car :

. Il ne répondait pas à toutes les exigences de la fiche de caractéristiques militaires (passage de fossér à bords francs, de troncs d'arbres d'un certain diamètre ...) ... bien trop sévères; l'eût-il fait que la cellule de l'hélicoptère n'aurait pas résisté !

. Sa masse était de 120 kg et cela réduisait encore légèrement l'autonomie de l'appareil (depuis, la masse maximum au décollage du PUMA est passée de 6,4 tonnes à 7,4 tonnes ... !).
Je pense que ces deux exemples sont significatifs.

22) <u>La conduite elle-même du programme</u> : du danger du renouvellement tardif d'un parc.

Les matériels en service ne pouvant être prolongés indéfiniment, ne serait-ce que pour une question d'efficacité militaire, une décision tardive de renouvellement d'un parc entraîne presque toujours une série de conséquences qui s'enchaînent inexorablement et dont les deux principales sont :

A) Des <u>délais insuffisants pour l'étude</u> et <u>la mise au point technique</u> :

L'urgence du besoin conduit en effet à passer rapidement à la phase d'expérimentation dite "tactique", laquelle perd une grande partie de son intérêt si le matériel n'est pas techniquement "opérationnel".
L'exemple que l'on peut donner dans ce domaine est celui du "drone" R 20, dérivé du CT 20, dont les campagnes tactiques nécessitaient toujours la présence de spécialistes du constructeur et débouchaient ... sur des mises au point techniques, ce qui a, finalement,provoqué l'échec du programme.

B) <u>L'adoption prématurée du matériel</u> :

Ce qui conduit à mettre dans les unités un matériel appelé encore à subir de nombreuses modifications.Or des modifications trop nombreuses réalisées sur des matériels affectés dans les unités coûtent très cher d'abord et entraînent ensuite de longues périodes d'indisponibilité opérationnelle.

En tout cas, tout facteur permettant d'avoir le plus tôt possible un matériel répondant au mieux au besoin, et donc ne nécessitant ni période de mise au point trop longue ni des modifications trop nombreuses, est à prendre aujourd'hui en considération. Parmi ces facteurs il faut mettre à mon sens en 1ère place la qualité du dialogue "Opérationnels - Ingénieurs".
Cela signifie que les ingénieurs ont également un certain nombre de responsabilités importantes dans le bon déroulement du dialogue et donc dans le succès ou l'échec d'un programme.

3) <u>Les responsabilités de l'Ingénieur</u>:

31) <u>Adaptation du "produit" au besoin militaire</u> :

Quel que soit le niveau auquel se situe l'Ingénieur, responsable de la production ou, au plus haut niveau, chargé d'une politique industrielle, sa responsabilité majeure et son souci constant doivent être à mon sens de réaliser le matériel le mieux <u>adapté possible au besoin militaire</u> exprimé :

- Au niveau de la réalisation par exemple, que faire d'une merveille technique inutilisable en campagne; je me souviens, il y a longtemps, d'un obusier de campagne triflèche, petite merveille mécanique, dont on ne pouvait pratiquement déployer les trois flèches que dans une cour de caserne car cela exigeait un sol rigoureusement plan ... !

- Au niveau d'une politique industrielle, il est nécessaire parfois, et c'est normal, de prendre en considération les possibilités d'exportations futures, les plans de charge des différentes entreprises ... etc, mais ces considérations annexes ne doivent pas conduire à des modifications des caractéristiques techniques du matériel telles que le besoin opérationnel serait moins bien satisfait.

32) <u>La maîtrise des coûts</u> :

En deuxième lieu, l'ingénieur a, bien entendu, une responsabilité directe et importante dans les coûts du matériel, et ce à tous les stades de la réalisation.

La décision de réaliser un programme ne doit pas être l'occasion pour l'ingénieur de lancer études et recherches pour réaliser un équipement meilleur, plus performant, ou simplement plus moderne, ... si les équipements existants donnent satisfaction.

De même, l'ingénieur doit en permanence afficher la vérité des coûts. D'abord parce que l'opérationnel n'aura pas obligatoirement les moyens de payer des surcoûts excessifs, mais surtout pour provoquer éventuellement une révision des exigences opérationnelles.
C'est bien dans ce domaine de la maîtrise des coûts que l'on voit apparaître, encore plus qu'ailleurs, la nécessité d'établir un dialogue permanent entre l'ingénieur et l'opérationnel.

33) <u>La tenue des délais</u> :

Enfin, l'ingénieur a la responsabilité et le devoir de tenir les délais.

Nous avons vu, quand nous avons parlé des responsabilités de l'opérationnel, les conséquences néfastes d'une décision tardive de renouvellement d'un parc ou d'une adoption prématurée d'un matériel. Toute augmentation des délais de réalisation ne fera qu'aggraver ces conséquences qui, outre une augmentation importante des coûts, peuvent conduire à la mise en place dans les unités d'un matériel qui nécessitera de nombreuses modifications pour être adapté à l'emploi opérationnel initialement prévu.

A la lumière des responsabilités réciproques de l'opérationnel et de l'ingénieur il apparaît bien la nécessité d'un dialogue fructueux entre ces deux personnages clés.
Des améliorations significatives de ce dialogue sont-elles encore possibles ? C'est ce que je vous propose d'examiner maintenant.

4) <u>Les améliorations du dialogue Opérationnels - Ingénieurs</u> :

Ce dialogue, en règle générale, est déjà de bonne qualité, mais ceci est dû souvent, en grande partie, aux liens privilégiés existant entre les partenaires. Ceci n'est plus suffisant pour assurer la réussite des programmes futurs.

Le dialogue opérationnels - ingénieurs doit être conçu, préparé, organisé, compte-tenu de son importance croissante, pour un maximum d'efficacité.
Tel qu'il existe aujourd'hui, il peut et doit impérativement être amélioré et dans la forme et dans le fond.

41) <u>Dans la forme</u> :

Il est fondamental que ce dialogue s'établisse effectivement dès la phase de <u>CONCEPTION</u> du <u>MATERIEL</u> qui est la plus importante à la fois pour l'adaptation de ce matériel à l'emploi opérationnel et surtout, aujourd'hui, pour le coût du programme. Il ne faut en effet jamais oublier que 65% des dépenses sont conditionnées par les décisions prises pendant la phase de conception, 80% pendant les phases de conception et de développement. (voir planche 2).

Bien entendu, c'est encore beaucoup plus vrai pour un programme exigeant la coopération de plusieurs pays.
Il est tout aussi important que ce dialogue s'établisse d'emblée à tous les niveaux concernés :

- Directions générales des programmes,

- Directions techniques,
- Organismes responsables des essais futurs ...
- Etc ...

C'est donc tout un dispositif d'ensemble qu'il s'agit de mettre en place et il faut accepter d'en payer le prix, comme l'on fait les différentes nations collaborant pour le programme de l'hélicoptère NH 90.
Il est d'ailleurs permis de s'étonner, alors que les militaires ont, dans le domaine opérationnel, une longue pratique des échanges d'officiers de liaison que les mises en place de personnels pour le suivi des programmes techniques importants se fassent toujours dans le même sens : Etats-Majors, vers Services Techniques.
La présence d'ingénieurs, détachés dans les Etats-Majors, ne serait peut-être pas inutile, ne serait-ce que pour saisir mieux le besoin opérationnel.

42) Dans le fond :

L'objectif dans ce domaine est d'aboutir à une meilleure compréhension réciproque et cela passe par une modification des comportements.

421. Meilleure compréhension:

C'est une question de formation, d'information et de stabilité des personnels.
Un effort important de formation des personnels a été consenti et l'Armée de Terre Française envoie depuis plusieurs années des officiers de haut niveau dans les plus grandes écoles d'ingénieurs pour les spécialiser dans les postes de responsabilités techniques et disposer ainsi d'officiers compétents capables de servir d'interlocuteurs aux ingénieurs et spécialistes des Directions Techniques. Cet effort est bien sûr à poursuivre. L'amélioration dans ce domaine est à rechercher d'abord dans une plus grande **STABILITE** des officiers en place dans les postes importants = responsable d'un programe, membre d'une équipe de marque ... sans que cela nuise au bon déroulement de leur carrière.
Il ne fait aucun doute que certains de ces officiers exercent des responsabilités aussi importantes pour l'intérêt général de l'Armée de Terre que le commandement d'un Corps de Troupe.
Les ingénieurs de leur côté doivent être informés concrètement sur les problèmes militaires, sur les besoins tactiques, sur l'environnement au combat, afin de bien comprendre ce que "veulent" les militaires : une centrale de cap destinée à un avion de ligne ne peut avoir les mêmes caractéristiques que celle destinée à un hélicoptère de combat appelé à effectuer en permanence des évolutions à grande inclinaison.

422. Modification des comportements :

Enfin, et ce sera le dernier point que j'évoquerai, il me paraît très important aujourd'hui d'aborder les problèmes d'équipement d'une Armée avec une mentalité différente de celle qui existait parfois dans le passé, et on rejoint là les problèmes de **FORMATION** de nos officiers et d'**INFORMATION** de nos ingénieurs.
- **Pour l'ingénieur**, il ne s'agit plus de répondre seulement au besoin opérationnel exprimé, sans le remettre en cause, en proposant ce qu'il y a de mieux et de plus cher. Il s'agit au contraire d'amener "l'opérationnel" à préciser mieux le besoin, à renoncer parfois à certaines exigences exorbitantes, puis de développer au moindre coût des équipements adaptés au mieux à l'emploi, en affichant toujours la vérité des coûts.
Le temps où l'on sous-estimait volontairement le coût d'un programme dans l'espoir de le faire adopter plus facilement est bien révolu.

- De même, pour l'officier d'Etat-Major, il ne s'agit plus de se contenter d'afficher les exigences idéales, en laissant le soin à l'ingénieur de trouver les solutions techniques adéquates.
Il s'agit au contraire d'avoir des exigences réalistes, aidé en cela par l'ingénieur, adaptées à un besoin tactique bien évalué, et pour cela, de bien situer ses exigences dans le concept général que l'on a retenu.
Cela me donne l'occasion de préciser les idées actuelles de l'ALAT Française dans le domaine de la réalisation d'hélicoptères de combat et de leur "**SURVIVABILITE**" sur le champ de bataille.

- Tout d'abord, doit-on réaliser des hélicoptères polyvalents ou spécialisés ?
C'est l'**hélicoptère SPECIALISE** qui est choisi, à la fois pour des questions d'efficacité opérationnelle et de coût.
En effet, un **hélicoptère POLYVALENT**, c'est-à-dire disposant d'équipements lui permettant de remplir plusieurs types de missions n'utilisera toujours qu'une partie de ses équipements. Cela conduit à transporter des charges inutiles et tout kilo d'équipement transporté augmente la masse de l'hélicoptère de 3 kg.(voir planche 3).
On aboutit donc à un appareil plus **LOURD**, donc moins discret, donc plus **VULNERABLE**.
Mais surtout, les différentes fonctions de combat (renseignement, protection, AC ...) doivent pouvoir être assurées au même moment, en des endroits différents du champ de bataille.

- S'agissant de la **VULNERABILITE**, ou plus exactement de la "**SURVIVABILITE**", le facteur le plus important nous paraît être d'abord la **DISCRETION**, ce qui est cohérent avec le choix d'un hélicoptère spécialisé et léger; en deuxième lieu l'**ALERTE** (dispositifs d'alerte laser, radar, IR ...).
Nous paraît également indispensable, en troisième lieu, la **PROTECTION**, c'est-à-dire le **BLINDAGE**, mais qui doit se limiter aux parties **VITALES** de l'appareil pour ne pas obérer trop la masse, donc la discrétion et les qualités de vol.

* **CONCLUSION:**

Les programmes d'armement sont devenus d'une telle complexité et exigent de tels investissements que "opérationnels" et "ingénieurs" sont de plus en plus liés par des intérêts communs.

Ceci implique que les attitudes suivantes :

- l'**Opérationnel** commande, l'**Ingénieur** réalise,

- ou (voir planches 4 et 5)

- l'**Ingénieur** propose, l'**Opérationnel** s'adapte,

ne sont plus de mise, si l'on ne veut pas aboutir à des incompréhensions graves et des surprises désagréables.

La réalisation des programmes d'hélicoptères armés futurs en particulier exige un dialogue intense et permanent entre l'opérationnel et l'ingénieur.

Tout cela pour dire que nous aurons de plus en plus besoin les uns des autres et nous ne pouvons que nous en féliciter.

PLANCHE 1

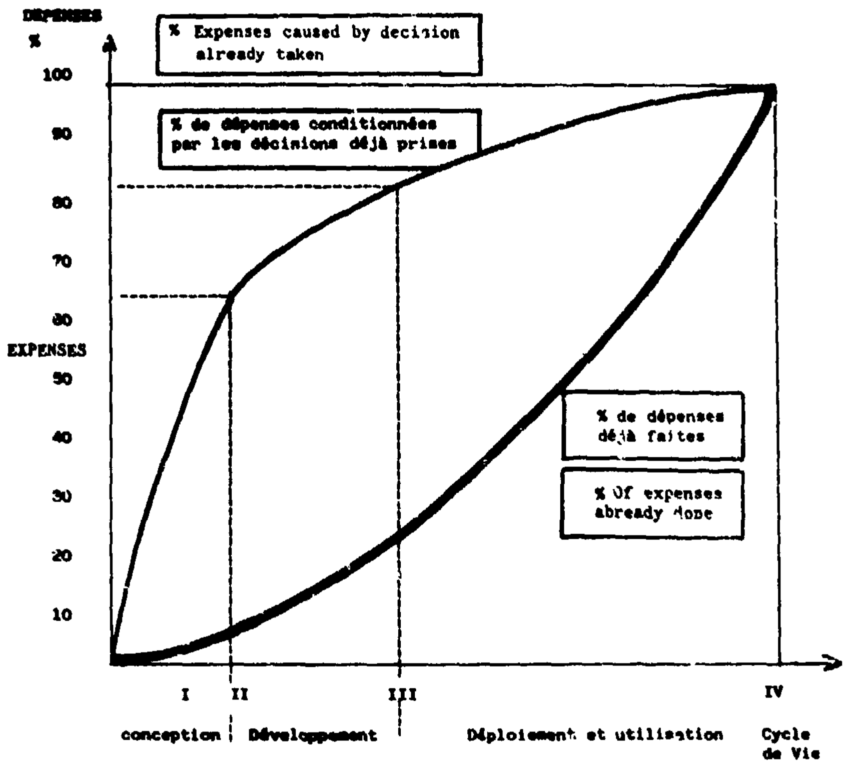
HELICOPTERE "PUMA" EQUIPE DU TRAIN AUTOMOTEUR DOP : POSSIBILITE DE CAMOUFLAGE.

PUMA SA 330 EQUIPED WITH SELF PROPELLED GEAR "DOP" POSSIBILITY OF CAMOUFLAGE

PLANCHE 2

EVOLUTION DES DEPENSES

ENGAGEES POUR UN PROGRAMME NOUVEAU



PROGRESS OF EXPENSES

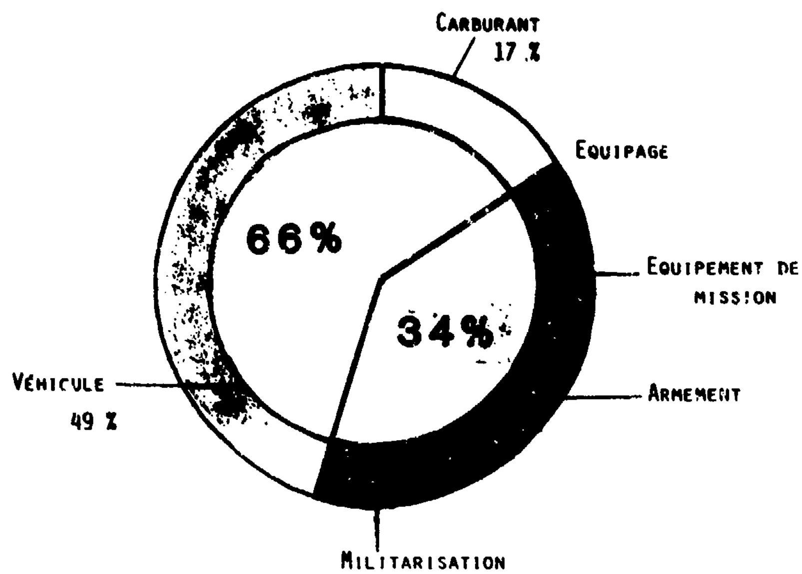
GENERATED BY A NEW PROGRAM

PLANCHE 3

REPARTITION DES MASSES

DANS UN HELICOPTERE ARME

DE 5 A 6 T



M	=	0,49M	+	0,17 M	+	m
MASSE DE MISSION		PORTEUR AIRFRAME		CARBURANT FUEL		EQUIPEMENTS EQUIPEMENTS

0,34 M = m

M ≃ 3 m

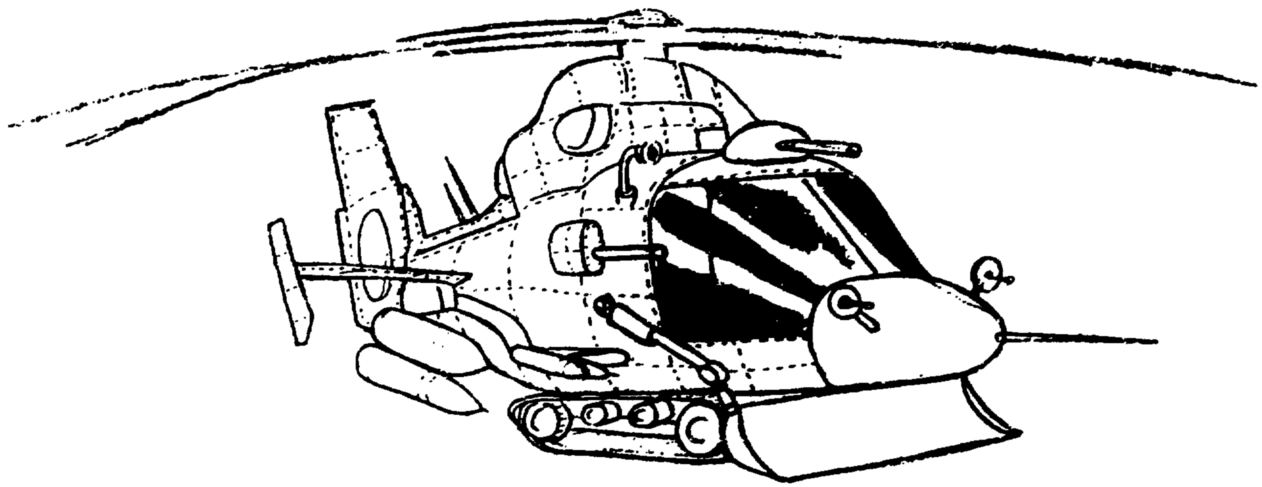
$$\boxed{\Delta M \simeq 3 \Delta m}$$

SPLIT OF WEIGHT

ON A 5 TO 6 T

ARMED HELICOPTER

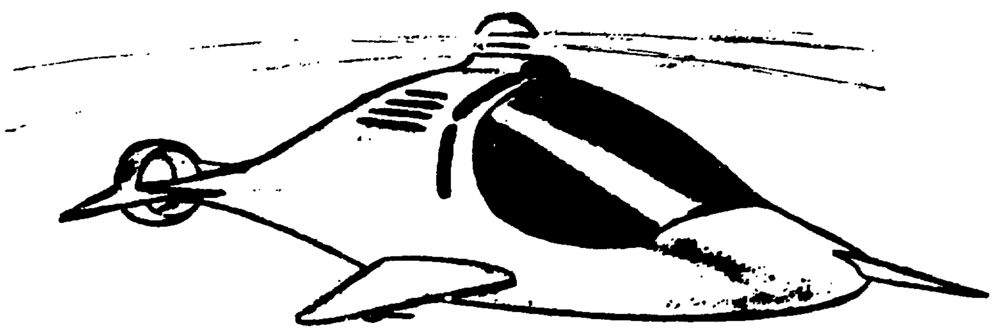
PLANCHE 4



CE QUE VEUT LE MILITAIRE : VITESSE + <u>ARMEMENT</u>

WHAT THE MILITARY WANT : SPEED + <u>WEAPONS</u>

PLANCHE 5



CE QUE RISQUE DE COMPRENDRE L'INGENIEUR : ARMEMENT + VITESSE

WHAT THE ENGINEER MIGHT UNDERSTAND : WEAPONS + SPEED

MILITARY-SCIENTIFIC DIALOGUE

by

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I should first like to say how honoured and happy I am to participate in this symposium, and I thank the organisers for allowing me to talk to such a learned gathering on a subject which I have for many years considered of paramount importance. It forms part of the much wider question of how to achieve the necessary conditions for the satisfactory execution of a new equipment programme. I should like to talk about the relationship which must be established between "tacticians" and "technicians", or, more precisely, of the military-scientific dialogue i.e. the dialogue between military staffs, and the procurement and technical managements.

One must of course recognise that this "dialogue" has always been the subject of much criticism, and this is still true today, albeit to a lesser extent. People often like to stress the differing preoccupations of the military staff and the procurement and technical management:

The former need to obtain an equipment which best meets the operational requirement,

For the latter their preoccupation is primarily to ensure a full work load in certain industries, or export sales, which sometimes results in the partial "overlooking" of the military requirement.

At another level, and in the habitual language of military programme managers, three comments crop up regularly:

— "It isn't what we wanted"
— "It's arrived too late"
— "It costs too much",

without going further into an analysis of where the responsibility lies for such a state of affairs.

These approaches are no longer acceptable, and the military-scientific dialogue must be continually improved, for the success or failure of an equipment programme can depend on the quality of this dialogue. This is particularly true in the case of helicopter procurement.

The basic elements of the dialogue have, in fact, been modified to a great extent in the recent past, giving it a quite new dimension.

I therefore propose to look at the following points:

— The new elements in the military-scientific dialogue,
— The responsibilities of operational staff in defining the equipment requirement and in programme management,
— The responsibilities of scientists in solving the technical problems,
— Possible improvements to the dialogue.

In doing this, I shall indicate to you the general direction in which French Army Aviation is heading in the field of equipment procurement.

1. NEW ELEMENTS IN THE MILITARY-SCIENTIFIC DIALOGUE

Three important new factors have recently greatly modified the elements of the military-scientific dialogue, giving it a new dimension.

The first concerns the process by which a new scientific technique is adapted to meet an operational requirement. Up to the middle of the Second World War, it was the operational requirement which led to or hastened scientific progress in a given area or direction: *the operational requirement preceded scientific progress*. For example, there is no doubt that it was the essential requirement to guarantee the security of the British Isles which lay behind the rapid progress made in the field of radar.

We now witness daily some "explosion" of discovery or scientific progress in many areas, and in numerous directions. It is thus no longer for the operational staff a matter of being content to state a requirement and then leaving to the scientist the responsibility of producing the technical answer; on the contrary, it is a question of the *operational staff and scientists together* perceiving, before any potential enemy does, technical advances which have the greatest relevance in the operational area.

The second factor concerns the construction of the instrument of combat itself. Up until now, at least in French Army Aviation, we have always adapted a weapon to an existing helicopter, one moreover designed for civilian use.

We have militarised civilian helicopters (with one exception: the SA 330 Puma, which possessed the necessary military characteristics, but it was not an armed helicopter). It is obvious today that we will only get a high performance combat helicopter if we *build the helicopter round, and as a function of, the chosen weapons system*. I will give just one example.

Having decided that one wants an anti-tank helicopter with either a nose-mounted, or a roof-mounted or a mast-mounted sight, it is not the same helicopter one builds in each case.

Finally, the third factor — and I have kept the best till last — is *the importance assumed by the cost* which can by itself, if it is badly managed or not managed at all, call into question the continuation of a programme.

2. THE RESPONSIBILITIES OF OPERATIONAL STAFFS

Let us then, in the light of these three factors, look at the responsibilities of the operational staff. These are exercised in two important areas:

— the statement of the Operational Requirement,
— the management of the programme itself.

In both these areas there are major pitfalls which must be avoided if, in the worst case, the programme is not to be compromised or, at best, the cost of the programme is not to be greatly increased. Here I am addressing an audience composed essentially of scientists and engineers; those who are often accused of producing equipment which costs too much. Therefore let me say quite clearly (and believe me it is not at all in order to please you) that the operational staff often share a large part of the blame for the excessive cost of equipment, either because the requirement has been badly appreciated or not sufficiently precise, or else because the programme itself has been badly managed.

2.1 The Appreciation and Statement of the Requirement (the danger of making exorbitant demands)

It would be impossible to overstate the importance of this first expression of the requirement which the "statement of military characteristics" constitutes (and which doubtless has its equivalent in other armies). It is a fundamental document; when drawing it up the operational staff must exhibit realism in not asking for either the useless or the impossible, and always remain forward-looking. How many projects have been still-born at the stage where the requirement is stated because the statement has been badly appreciated as regards the operational need and technical possibilities, and on which nonetheless large sums of money have been expended? Here are two significant examples:

— The first is the so-called "complete" version of the SA 330 Puma, which was to take off and land in conditions of zero visibility and cloud ceiling, without any ground-based aids, be capable of flying in cloud and icing atmospheric conditions, either alone or in formation groups, and so on (military characteristics statement dated 15.7.63). All this led to a considerable research effort and to the construction of completely new prototype equipments (the SAIGA radar obstacle detector, a station-keeping system etc.) but the helicopter never saw the light of day in the required time frame.

— The second example is more interesting, because it responded to an important operational requirement which could have been met by the proposed equipment if the required performance characteristics had been more realistic, and if the operational staff had possessed a better knowledge of the evolution of the helicopter. I refer to the "DOP self-propulsion undercarriage" which was studied for the SA 330 Puma (Figure 1). The aim was to enable the helicopter to move short distances on rough ground to take advantage of nearby cover or shadow in order to improve its camouflage. An SA 330 (06) was therefore fitted with an ingenious device (a rack-type auxiliary propulsion unit powered by one of the helicopter's two hydraulic circuits). This was not accepted because:

  — It did not meet all the military characteristic requirements (e.g. ability to negotiate steep-sided ditches, fallen tree trunks of a certain diameter, etc.) which were too exacting. The structure of the helicopter itself would probably not have survived such treatment!

  — Its weight was 120 kg, and that again slightly reduced the helicopter's range (since then the maximum take-off weight of the Puma has risen from 64 to 7.4 tonnes!).

2.2 The Managing of the Programme Itself (the dangers attendant on the tardy replacement of a major equipment)

As it is not possible to prolong the in-service life of an equipment indefinitely, if only on the grounds of military efficiency, a late decision to replace a major equipment almost always results in a series of consequences which are inexorably linked. The two principal ones are:

(a) *Insufficient time for study and technical adjustments*
The urgency of the requirement produces a rapid progression to the so-called "tactical" experimentation phase, which loses much of its relevance if the equipment is not technically operational. One can cite here the example of the R20 drone, a derivative of the CT 20, whose tactical trials invariably required the presence of specialists from the manufacturers, and which led to technical adjustments and in the end resulted in the failure of the programme.

(b) *Premature acceptance of an equipment*
This leads to equipment being introduced into units while still being the subject of numerous modifications. Now, to have too many modifications carried out on equipments already in service is very costly and, moreover, results in long periods of operational non-availability.

Any factor which allows us to obtain as quickly as possible an equipment which best meets the requirement, and thus needs neither a lengthy development phase nor too many modifications should be taken into consideration. In my view, pride of place among these factors should go to the quality of the military-scientific dialogue. This implies that scientists and engineers also have a certain number of important resposibilities for the proper conduct of this dialogue and thereby for the success or failure of a programme.

3. THE SCIENTIST'S RESPONSIBILITIES

3.1 Adapting the "Product" to the Military Requirement

Irrespective of the level of involvement of the scientist or engineer, whether he is responsible for production or, at the highest level, industrial policy-making, his main responsibility and constant concern must, to my mind, be to produce the equipment *best suited to the military requirements*.

– At the design level for example, what is the point in having a technical marvel which is unuseable in the field? I recall seeing, a long time ago, a tripod mounted howitzer which was technically a little beauty, but whose three legs could in practice only be deployed on a barrack square because they required a perfectly flat surface!

– At the level of industrial policy-making, it is sometimes necessary, and quite normal, to take into account overseas sales possibilities in the future, keeping various industries working at full capacity, and so on. However these secondary considerations must not lead to modifications of the technical characteristics of equipments to the extent that the operational requirement would be less well met.

3.2 The Mastery of Costs

The scientist or engineer has also, of course, a direct and important responsibility for an equipment's costs, and this is true at all stages of its procurement. The decision to carry out a programme must not be construed by the scientist as an opportunity to launch studies and research in order to produce an equipment which is better, has a superior performance, or is simply more modern, if existing equipments are perfectly satisfactory. Similarly, the scientist must continually point out the reality of costs. Firstly, because the military will not necessarily have the means to pay for excessive cost overruns, but above all to provoke a possible reconsideration of the operational requirements. It is in this area of keeping costs in check, more than in other areas, that one sees just how necessary it is to establish a permanent dialogue between operational staffs and scientists.

3.3 Keeping Programmes on Schedule

It is also the duty of the scientist or engineer to keep programmes on schedule. We saw, when speaking of the responsibilities of the operational staff, the damaging consequences of late decisions to replace an equipment, or the over-hasty acceptance into service of an equipment. Any delay in procurement can lead to a large increase in cost, and an introduction into service of an equipment which will require numerous modifications before it can be used operationally in the way which was originally foreseen. In view of the shared responsibilities of operational staffs and scientists, the necessity for a fruitful dialogue between these key personalities becomes quite clear. Are any further improvements to the dialogue possible? That is the question which I now propose to address.

4. POSSIBLE IMPROVEMENTS TO THE MILITARY-SCIENTIFIC DIALOGUE

As a general rule this dialogue is already of a high quality, but this is often in large measure due to the privileged contacts which exist between the two sides. This alone is not sufficient to ensure the success of future programmes. The military-scientific dialogue must be conceived, prepared and organised, given its growing importance, to attain maximum efficiency. As it exists today it can and must be improved as regards both its form and its substance.

4.1 The Form

It is fundamental that this dialogue be effectively established during the EQUIPMENT DESIGN phase, which is the most important, both for adaptation of the equipment to operational use and, today, for the cost of the programme. Indeed, one must never lose sight of the fact that 65% of costs are dependent on decisions taken during the design phase, and 80% on those taken during the design and development phase (Figure 2). This is of course especially true for collaborative projects. It is equally important that this dialogue be established from the outset at all levels including Programme Management, Technical directorates, Trials organisations, etc.

It is thus a matter of setting up a comprehensive system, and one must be prepared to pay the price of all that this implies, as did the various nations collaborating in the NH 90 helicopter programme. Whereas the military have for many years taken as normal the practice of exchanging liaison officers for operational purposes, one might understandably be surprised that the detachment of personnel to liaise on technical programmes is always done in the direction *from* military staffs *to* technical services. The presence of scientists attached to military staffs would perhaps serve a useful purpose, were it only to promote a better understanding of the operational requirement.

4.2 The Substance

The aim here is to arrive at a better reciprocal understanding, and that requires a change in attitudes.

4.2.1 A Better Understanding

This is a question of the education, information and continuity of the personnel involved. For several years now the French Army has made a considerable effort in the scientific education of officers, and has posted high quality officers to scientific centres of excellence to prepare them for technically accountable posts. It thus has a pool of competent officers capable of engaging in dialogue with civilian scientists and engineers of the technical directorates. This effort must of course be sustained. Any improvement in this area must first be sought through greater CONTINUITY of officers in important posts i.e. programme managers, or team members in particularly important programmes, without adversely affecting their careers. There is no doubt that some of these officers bear responsibilities every bit as important to the interests of the service as do field commanders. For their part, scientists and engineers must be well informed in a practical way on military problems, on tactical

requirements, and on the battlefield environment in order to appreciate properly what the military need. For example, a heading reference system designed for a civil passenger aircraft will not necessarily have the same characteristics as one which is to be used in a combat helicopter continually required to manoeuvre to extreme attitudes.

4.2.2 A Change in Attitudes

Lastly, it seems to me very important nowadays to address the problem of army equipment procurement with a different mentality to that which sometimes applied in the past. This brings us back to the questions of EDUCATING our officers and INFORMING our scientists and engineers.

For the scientists or engineer it is no longer a matter of responding unquestioningly to the operational requirement, as framed by the military, and merely proposeing the best and most costly solution. On the contrary, his task is to get the operational staff officer to spell out his requirement as precisely as possible, to get him to occasionally relax certain exorbitant demands, and then to develop at minimal cost the equipment which best suits his purpose, all the while pointing out the financial realities. The time when one deliberately underestimated the cost of a programme in the hope of getting it adopted more easily is long since past.

In the same way the staff officer can no longer be content to simply express Utopian requirements and then leave it to the scientist to find suitable answers. On the contrary he must make realistic demands, helped in this by the scientist; demands which are appropriate to a well-reasoned tactical requirement, based on the current overall general concept.

This provides me with the opportunity to say a few words on current thinking in the French Army Aviation on the procurement of combat helicopters and their SURVIVABILITY on the battlefield.

First of all, ought one to construct multi-purpose or specialist helicopters? It is the SPECIALIST helicopter which has been chosen, on the grounds of both operational efficiency and cost. In fact a MULTI-PURPOSE helicopter (i.e. carrying a number of equipments which enable it to carry out several types of mission) will invariably use only a proportion of these equipments. That means it is carrying extra weight needlessly, and every kilogram carried increases the weight of the helicopter by 3 kg (Figure 3). One ends up with an aircraft which is HEAVIER, less discreet, and therefore more vulnerable. But above all, the different combat functions (intelligence gathering, protection, anti-armour etc.) must be performed at the same moment, at different places on the battlefield, which is an obvious impossibility.

On the question of VULNERABILITY or, more precisely, SURVIVABILITY, the most important factors seems to us to be firstly the helicopter's DISCRETION, which explains the choice of a light, specialist helicopter; and, secondly, WARNING (laser, radar, infra-red, warning devices etc.), and thirdly, PROTECTION, that is to say ARMOUR. This armour should be limited to the VITAL parts of the aircraft to keep its weight down and thus maintain discretion and flight characteristics.

CONCLUSION

Weapons systems procurement programmes have reached such a degree of complexity, and demand such heavy investment, that operational staffs and scientists are increasingly bound by common interests.

This implies that neither of following attitudes whereby:

— The military demands and the scientist produces (Figures 4 and 5)
— The scientists proposes and the military adapts,

are acceptable any longer if one wishes to avoid serious misunderstandings and unpleasant surprises. In particular the satisfactory execution of future armed helicopter programmes will necessitate a close and permanent dialogue between the staff officer and the scientist. All of which proclaims that we shall depend on each other more and more, and we must congratulate ourselves on that.

THE INFLUENCE OF OPERATIONAL REQUIREMENTS ON LHX CONCEPT FORMULATION

BY
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SUMMARY

This paper addresses the U.S. Army's Light Helicopter Family (LHX) helicopter program which is currently in concept formulation. The paper describes the activities associated with concept formulation including governing regulations, how the process is executed, the scope of effort involved and innovations included for LHX. The evolution of the requirements is emphasized and the focus of the paper is on how operational requirements drive engineering requirements and ultimately the design of the aircraft. Operational requirements are driven by the LHX concept, namely, a family of Scout/Attack (LHX-SCAT) and Utility (LHX-U) aircraft with common dynamic components, core mission equipment and common subsystems. The impacts of the unique LHX-SCAT aircraft on the design and commonality with the LHX-U are discussed. The operational requirements are categorized into those associated with the Army 21 concept of the future battlefield, the threat, safety and Reliability, Availability and Maintainability (RAM). Each category results in several specific design impacts. The synergistic effect of these design impacts creates a very challenging task when combined with real world cost, weight and size goals for the LHX.

LIST OF ACRONYMS

ABC - Advancing Blade Concept
ACAP - Advanced Composite Aircraft Program
AH-64 - Advanced Attack Helicopter (APACHE)
AMC - Army Materiel Command
AAMAA - Army Aviation Mission Area Analysis
ARTI - Advanced Rotorcraft Technology Integration
BTA - Best Technical Approach
COEA - Cost and Operational Effectiveness Analysis
DA - Department of the Army
DOD - Department of Defense
EO-TADS - Electro-Optical Target Acquisition and Designation System
FSD - Full Scale Development
JMSNS - Justification for Major System New Start
LHX - Light Helicopter Family
LHX-U - Light Helicopter Family - Utility Helicopter
LHX-SCAT - Light Helicopter Family - Scout/Attack Helicopter
LOA - Letter of Agreement
MANPRINT - Manpower, Personnel and Integration Training
MEP - Mission Equipment Package
NBC - Nuclear Biological Chemical
NOE - Nap-of-the-Earth
O&O - Operational & Organizational
PMGW - Primary Mission Gross Weight
RAM - Reliability, Availability, and Maintainability
ROC - Required Operational Capability
SDGW - Structural Design Gross Weight
TOA - Trade-Off Analysis
TOD - Trade-off Determination
TRADOC - Training and Doctrine Command

WHAT IS LHX

The LHX will consist of two advanced technology conventional helicopters, a Scout/Attack (SCAT) and a Utility (LHX-U) version with common dynamic components (engine, rotors, drive system), common core Mission Equipment Package (MEP) and common subsystems as depicted in Figure 1.. The SCAT will be designed to autonomously perform both the Scout and Attack helicopter functions, e.g., find and destroy enemy targets. The LHX-U will perform light utility helicopter missions; resupply, team insertion and command, communication and control. The LHX will replace our current light fleet of rapidly aging and tactically obsolescing UH-1, AH-1, OH-6 and OH-58 series of helicopters and complement our heavy fleet of AH-64 attack and UH-60 utility helicopters.

WHY LHX

As detailed in reference 1, the initial need for the LHX was established as a result of the Army Aviation Mission Area Analysis (AAMAA) which identified the current light fleet deficiencies depicted in Figure 2. These deficiencies result from the specific condition of the Army's current light fleet as shown in Figure 3. The economic and logistics supportability problems arise from the profileration of the current light fleet aircraft, the age of these aircraft and the fact that these aircraft incorporate 1950's technology. They obviously are lacking in terms of countering the current or future threat and also are severely deficient in addressing the expanded battlefield of the 1990's in terms of high altitude, hot day operation.

CONCEPT FORMULATION

The LHX is currently in the concept formulation phase, which immediately precedes Full Scale Development (FSD). Concept formulation is guided by AR-1000-1, Basic Policies for Systems Acquisition and AR 71-9, Material Objectives and Requirements. Concept formulation was initiated by approval of the Justification for Major System New Start (JMSNS) in 1982. This one page document describes the concept to be explored, why a new system is required and establishes initial major goals for the program. The next step is for the Army's Training and Doctrine Command (TRADOC) to define how the helicopter will be utilized and what types of missions it will perform. This is documented in the Operational and Organizational (O&O) plan. The LHX O&O plan was guided by the US Army's projections of what the future battlefield will be like around the year 2000, officially called the Army 21 concept. This formulation process is set up to systematically determine specific requirements and prepare for development of the system. The requirements process, starting with very general requirements in the O&O plan, evolves through the Letter of Agreement (LOA) with more specific but flexible requirements into the Required Operational Capability (ROC) with generally specific, firm requirements. While the requirements process is primarily the responsibility of TRADOC, the Army Materiel Command (AMC) which has the responsibility for developing and procuring systems which meet TRADOC's requirements, plays a major role in advising and providing data to assist in the requirements evolution.

The concept formulation process as depicted in Figure 4 consists of a Trade-Off Determination (TOD), prepared by AMC, a Trade-Off Analysis (TOA), prepared by TRADOC, a Best Technical Approach (BTA), prepared by AMC and a Cost and Operational Effectiveness Analysis (COEA), prepared by TRADOC. Each command (i.e., TRADOC and AMC) supports the other in supplying data, review and coordination of each of the aforementioned documents, and in the case of the LHX program, the US helicopter industry also had substantial involvement in the process, primarily through three contracts; LHX Preliminary Design, LHX Wind Tunnel/Simulation and the Advanced Rotorcraft Technology Integration (ARTI) programs.

The TOD process as outlined in Figure 5 provides data on candidate aircraft and subsystems along with technology descriptions, cost, weight, technical risk, and Reliability, Availability and Maintainability (RAM) implications. The TOD provides the first set of preliminary designs of candidate systems which can meet the general requirements of the O&O plan. Sets of designs are provided to ensure a full range of options, in terms of specific requirements, are available since firm requirements are not known at this stage of concept formulation.

For example, in the LHX TOD designs were formulated for five configurations; conventional helicopters, compound helicopters, Advancing Blade Concept (ABC) helicopters, compounded ABC helicopters and tilt rotor aircraft. During the TOA the TOD outputs, including preliminary designs, are examined using limited effectiveness modeling and substudy analysis to begin the process of translating broad concepts into specific requirements. Normally the primary product of the TOA, in addition to the TOA report, is the first specific requirements document, the LOA. The LOA is an agreement between AMC and TRADOC which allows planning, to include program development funding, for FSD. The LOA contains specific, but flexible preliminary requirements, which allows AMC to formulate the BTA. Thus, the BTA can be prepared based upon specific requirements which allows for specific design candidates as opposed to the multitude of design variations in the TOD. For example, for the LHX BTA, the only configuration included was the conventional helicopter. However, US Army regulations dictate that a new development system cannot be assumed a priori, and thus design derivatives of existing military and commercial aircraft must be included for analysis in the next step in concept formulation, namely the COEA. In the COEA, the BTA's (new development and derivative designs) are thoroughly examined utilizing sophisticated force-on-force effectiveness analysis (i.e., war gaming) to provide definite data on the effectiveness of various fleets (fleets are formulated for each set of alternative airvehicle systems). The results of the COEA, coupled with military judgement, are used to provide Department of the Army (DA) and Department of Defense (DOD) senior management with quantifiable comparisons of alternatives and also to define the most critical pre-development document, the ROC. Although the previous description of the concept formulation process seems rather straightforward and rigid, in actual practice a tremendous amount of negotiation between AMC and TRADOC takes place along the way and in the case of the LHX, industry has also been intimately involved, along with high level DA officials.

PRELIMINARY DESIGN PROCESS

The evolution of operational concept to operational requirements to technical requirements culminating in impacts to formulation of preliminary designs is depicted in Figure 6, the preliminary design process. This iterative process takes place throughout concept formulation with design changes directly related to the evolution of requirements and also fluctuating somewhat with changes in available technology.

The preliminary designs described in this paper are Government designs; and as such, may not be representative of the actual LHX, which will of course be designed by industry, not by the Government. However, a substantial effort is placed on formulation of the Government designs, since they do play a significant role in the concept formulation process.

As noted previously, the first LHX designs were formulated during the TOD, and at this point in the program, several major goals and the LHX concept had been established and essentially remain the same today. The LHX concept and major program goals significantly effect the LHX preliminary designs and will be discussed first to better understand how operational requirements influence the designs. The program goals and concepts which affect the design include: Combining scout and attack functions in the SCAT, a SCAT Primary Mission Gross Weight (PMGW) of 8000 ± 500 lbs. (3628.8 ± 226.8 Kg), Utility dynamic components (engines, rotors, transmission) common with SCAT, average unit flyaway cost of

$6M/SCAT and $4M/Utility, built in growth capability (weight and performance) and single pilot SCAT and single pilot operable Utility, all consistent with the overall LHX concept of small, lightweight and affordable. In addition, during the TOD the decision was made to develop an advanced technology, lightweight engine in order to help attain weight (through fuel efficiency), reliability and operational and support cost goals. The engine development program was initiated in 1983 in order to ensure availability of Government qualified engines for LHX FSD. A twin engine configuration was chosen for safety and service commonality. To provide built in growth potential the T800 engine will be designed for 1200 shp (894.8 kw).

The SCAT concept means that both the Scout and Attack missions as shown in Figure 7 are performed by a single aircraft and dictates that mission equipment for acquiring and tracking targets (i.e., OH-58D scout) and armament to kill targets (i.e., AH-64) must be combined in a single aircraft. This translates into approximately 1600 lbs. (725.8 Kg) for mission equipment and armament.

The SCAT weight goal(which is a goal as opposed to an absolute limit) is a primary driver in terms of levels and types of technology required for the LHX. Weight, maintainability, battle damage tolerance and repair dictates maximum use of composite materials for the LHX system. Thus, the LHX airframe will be an all composite design tailored from the experience learned from results of the Advanced Composite Aircraft Program (ACAP) demonstration vehicle designs. The rotor blades and rotor hubs will also be made of composite materials. In addition, all subsystems will consider composite materials application where practical for the overall weight goal to be achieved. Gun barrels, drive shafts and generator housings are examples of subsystems which may utilize composite materials to minimize aircraft weight. The LHX family concept calls for SCAT and Utility common dynamics and since the SCAT is the more complex aircraft with more demanding performance requirements, the Utility dynamic system will "fallout" from the SCAT design. The average unit flyaway cost goal complicates the design process which must assure trade-offs between weight, performance and acceptable costs. Tradeoffs involving cost, weight and performance have played an important role in the requirements process. The "built in" growth potential has also affected preliminary designs. As mentioned previously the T800 1200 shp (894.8 kw) engine is currently in development and twin T800 engines provide power in excess of that required to meet current performance requirements. The other half of allowing for "built in" growth requires additional structural integrity and aerodynamic capability. This has been addressed in the design process by specifying a Structural Design Gross Weight (SDGW) well in excess of the PMGW and including performance requirements (vertical rate of climb and maneuverability) at the SDGW. Thus, even for the stringent hot day design conditions, the LHX will have margins in engine power, aerodynamic capability and structural integrity as shown below:

	TAKEOFF POWER	ROTOR THRUST	STRUCTURAL LOAD FACTOR
AVAILABLE (100% IRP)	1692(1262 kw)	2.2g	4.1g
REQUIRED (PMGW)	1317hp (982kw)	1.9g	3.5g
MARGIN	28%	16%	17%

*500 ft/min (227m/min) VROC

The final goal/concept to be addressed is the single pilot goal for the SCAT configuration. Incorporation of the single seat contributes appreciably to meeting the system weight and cost goals. Specifically, 600-900 lbs. (272.2 - 408.2 kg) are saved compared to a two pilot configuration, resulting in an appreciable cost savings, even though the airframe is less than half of the total unit flyaway cost. Perhaps surprisingly, the single pilot goal is not a driver in terms of required mission equipment. The SCAT concept (autonomous scout and attack functions), threat driven timelines and the onboard weapons systems dictate mission equipment consistent with single pilot operation. The sophisticated mission equipment package along with a digital flight control system and integrated cockpit provides automated functions which should reduce workload allowing single pilot operation. A detailed explanation of this would require a separate paper; and therefore, will not be addressed further.

It is evident that even before the operational requirements are considered the LHX concept and program goals have fixed a significant number of design parameters; the number and type of engine(s), MEP and armament suites, SDGW, crew configuration and dynamic components of the Utility are established as a fallout of the SCAT design.

OPERATIONAL REQUIREMENTS

Operational requirements can be divided into four broad categories; Army 21 Concept, which projects the characteristics of the battlefield around the year 2000, Threat, Safety and Reliability, Availability, Maintainability (RAM). Each of these categories will then be expanded into system characteristics and the design impact for each system characteristic will be described.

ARMY 21 CONCEPT

The Army 21 concept of the future battlefield has three major thesis; (1) the battlefield will be dispersed and intense, (2) the US Army must be prepared to fight anywhere in the world and (3) we must be ready to fight around the clock regardless of environmental conditions. The dispersed battlefield implies that large areas must be covered and thus long range and high endurance are required. The design impact on the LHX is a requirement for a large internal fuel capacity and emphasis on efficient engines and rotor system. To put this in perspective, the current design has a 1600 lb. (725.8 Kg) internal fuel tank which combined with the fuel efficient T800 engine provides a combat range of over 300 n.m. and a mission endurance of 2.5 hours.

The need to be able to fight anywhere in the world translates into a significant hot day performance capability, and both self-deployment and transportability requirements. The LHX will be designed to perform all its missions with required armament and fuel loading at 4000 ft 95°F density altitude. This results in a relatively large diameter 40 ft (12.2m), main rotor for such a small aircraft. At PHGW this rotor provides a vertical rate of climb of almost 3000 ft/min (907 m/min) for the hot day condition and provides hover out of ground effect takeoff at gross weights 22% above PHGW. An even larger diameter main rotor might be preferred but the requirement to load as many aircraft as possible in a C-141 transport aircraft restricts larger diameter rotors since the rotor diameter also drives the overall fuselage length and the rotor/fuselage vertical separation required is a function of rotor diameter. Since airlift will be at a premium during a time of emergency, the Army wants to ensure that the LHX can get to the battlefield. Self-deployment capability will also be required for both the Northern and Southern routes to Europe. Air transportability also restricts the overall height of the aircraft thus complicating placement of mission equipment sensors and introducing potential aerodynamic interference complications due to decreased rotor/fuselage separation as was experienced during development of the Army's UH-60 and AH-64. Self-deployment will be accomplished with the addition of external fuel tanks which adds weight to the basic aircraft for structural provisioning and also, since the self-deployment mission will take over 8 hours, system reliability is a key factor.

Being ready and able to fly and fight around the clock at night and in adverse weather, enhances the need for a pilotage system to assist the pilot in night Nap-of-the-Earth (NOE) operations and an advanced state-of-the-art Electro-Optical Target Acquisition System (E-O TADS) to ease the burden of target acquisition to the maximum extent practical. These requirements add appreciable system cost and approximately 350 lbs. (158.8 Kg) to aircraft empty weight and present a design challenge in terms of sensor performance and placement to ensure unrestricted large field-of-view for single pilot operation and exceptional targeting capability necessary to counter future threats.

THREAT
Although the threat to LHX is encompassed in the Army 21 concept, threat will be addressed separately by consideration of both ground and air systems. In addition to the normal ground forces, the LHX is expected to engage in air-to-air combat with threat helicopters. This requirement, in addition to the obvious need for air-to-air weapons (turreted gun and air-to-air missiles) dictates at least speed parity with the threat system and significant maneuverability and agility capability. The need for increased speed is reflected in the LHX design innovations for low drag; namely, narrow fuselage (aided by single pilot design), retractable landing gear, engines essentially buried in the fuselage, high speed airfoils and tappered/swept tip main rotors and internal/conformal weapons pods. The twin T800 engines coupled with the above design characteristics allow the LHX SCAT to meet the 170 knot cruise speed requirement. For maneuverability, a high solidity rotor will be required to meet a high speed load factor capability at SDGW. Since results of air-to-air combat studies indicate that the ability to turn rapidly has benefits in both avoiding and engaging enemy targets, a high turn rate requirement for the LHX dictates both a large diameter and high solidity anti-torque/directional control system.

The vast numbers and increasing sophistication of threat ground acquisition systems and weapons makes the amount of time the helicopter is exposed (as opposed to masked by terrain or vegitation) an extremely critical parameter. To minimize exposure time, the system must be able to rapidly unmask, acquire and/engage targets and rapidly return to mask. Rapid acquisition of targets translate into a large search volume with an ability to scan rapidly and high speed data processing. LHX designs will require automated flight modes, such as automatic hover hold and automatic "bob up"to aid single pilot operation and minimize exposure time. The future battlefield will certainly include various low and medium caliber weapons from ground systems and ground personnel which requires the addition of ballistic armor for crew and critical component protection, as well as redundancy in flight controls and other critical components. Ballistic protection adds over 300 lbs. (136 Kg) to the aircraft empty weight.

Finally, the future battlefield is projected to have Nuclear, Biological and Chemical (NBC) threats. As in current systems, the LHX crew will wear protective NBC ensembles; but in addition, the LHX SCAT will have a sealed-pressurized cockpit and integral micro-climatic cooling system to provide additional protection and crew comfort. Also, both the SCAT and Utility will be designed to minimize internal and external crevices and open compartments to aid in the decontamination process.

SAFETY
The next two categories of operational requirements, Safety and Reliability Availability, and Maintainability are equally important for peacetime and wartime operation. As part of the TOD and TOA processes significant safety related requirements have been identified. These include; twin engines, crashworthiness protection equal that found in the much larger AH-64 and UH-60 aircraft, wire strike protection, frangible main rotor tips, external protection for the anti-torque system, and automated cockpit for workload reduction.

RAM
Finally, recognizing that system performance is only useful if the system is available when needed and that support resources are at a premium, special emphasis has been placed upon the LHX RAM characteristics. Good RAM characteristics can only result if actions are taken early in the design process. In the case of LHX, AMC has established an initiative, labeled Manpower and Personnel Integration (MANPRINT) to ensure RAM, safety, and training and human factors are thoroughly considered in the design process. Some of the specific impacts of this initiative on the current LHX design regarding RAM characteristics are as follows: Increased use of high reliability components,

self-healing architecture, onboard diagnostics and testability, redundant flight control and other critical systems, flight data recorders, designed in accessability for servicing, commonality of parts between the SCAT and Utility in addition to an overall reduction of parts through application of advanced technology design and fabrication.

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ACKNOWLEDGEMENT

The authors acknowledge the support of Dr. Michael Scully, Army Research and Technology Activity, NASA Ames Research Center and his staff who are responsible for formulating the LHX designs reported in this paper.

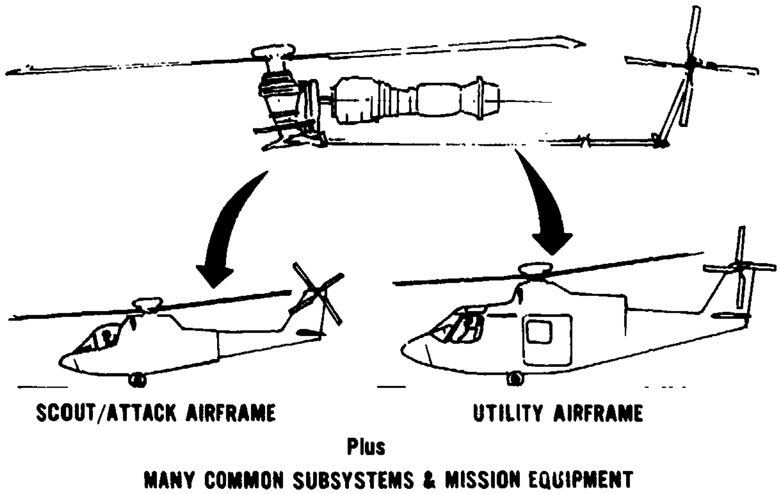


FIGURE 1. LHX COMMONALITY

FIGURE 2. AAMAA DEFICIENCIES

- Average Age Will Exceed 30 Years by Year 2000
- Deteriorating Economic and Logistic Supportability
- Insufficient Quantities to Satisfy Aviation Modernization Plan Force Structure
- Void of NBC Protection
- Insufficient Payload/Range for Air Land Battle 2000
- High Fuel Consumption in Comparison to Modern Engines
- Lack of Adverse Weather Capability (Rain/Fog/Obscurants)
- Inadequate Air-To-Air Capability
- Inadequate Against Advanced Threats
- Cumbersome Transportability and Not Self-Deployable

FIGURE 3. EXISTING LIGHT FLEET CONDITION

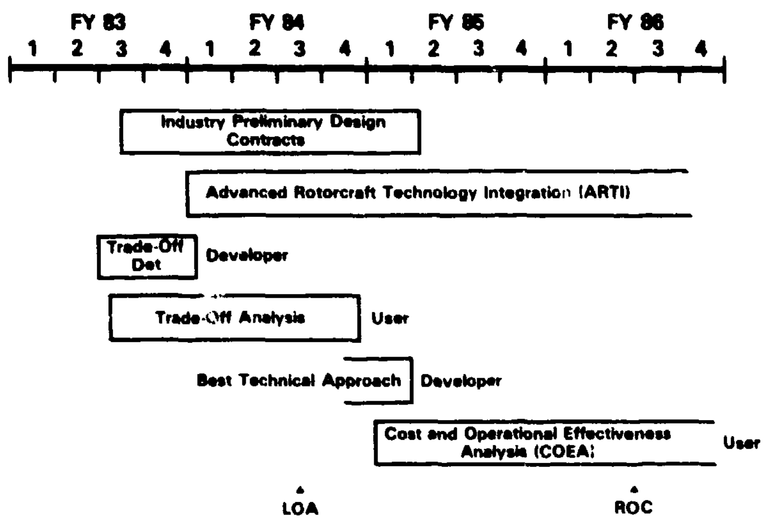
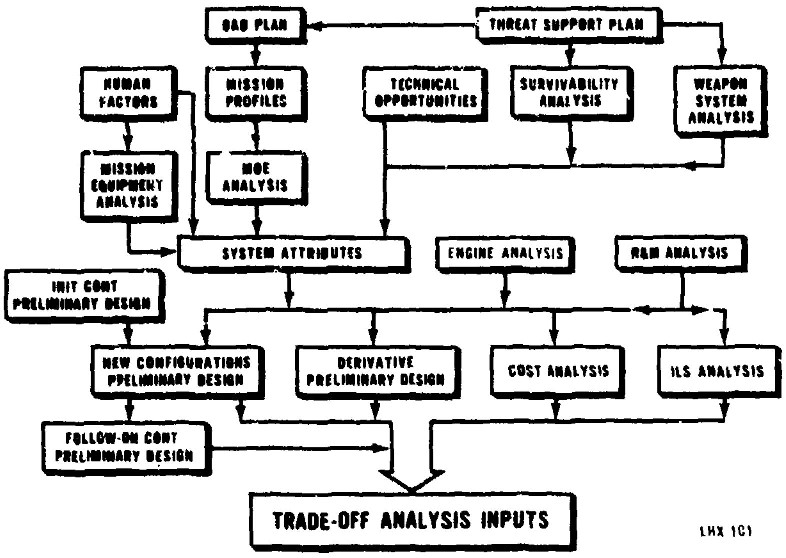


FIGURE 4. LHX CONCEPT FORMULATION



LHX 101

FIGURE 5. TRADE-OFF DETERMINATION

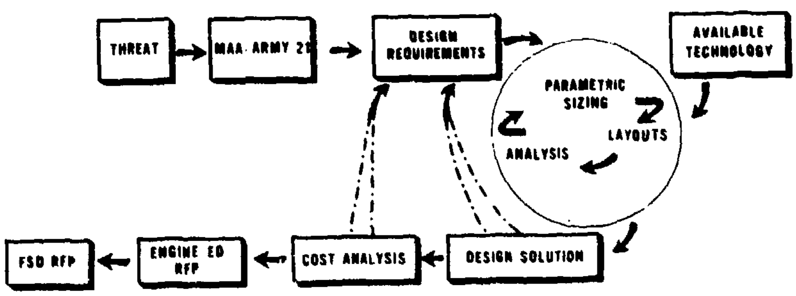
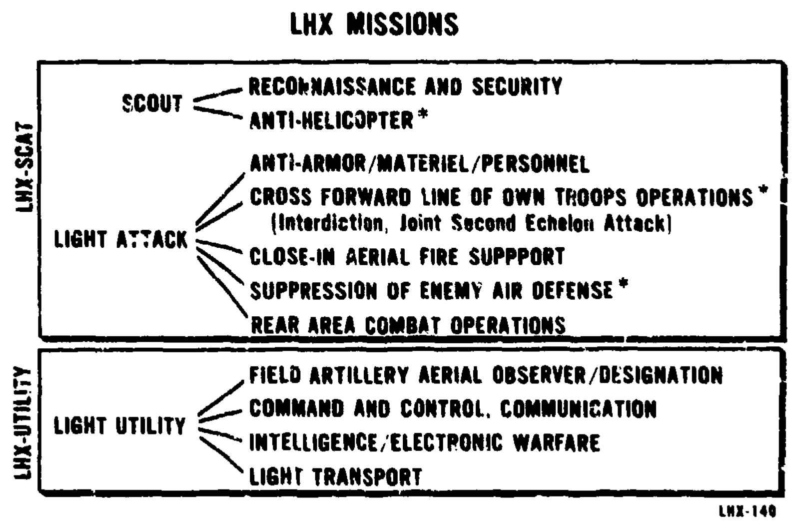


FIGURE 6. THE PRELIMINARY DESIGN PROCESS

LHX-268



* New Missions Required in Army 21

FIGURE 7. LHX SCAT MISSIONS

DESIGN REQUIREMENTS FOR FUTURE COMMERCIAL OPERATIONS

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ABSTRACT

This paper focuses on design requirements stemming from civil operations.
The offshore oil and gas industry is and will continue to be the main application of civil helicopters.
Development of other commercial market will require major improvements in almost all areas. Some of these areas are highlighted here: safety, reliability, cockpit ergonomics, vibration for example.
The current rate of progress is considered much too slow and concern is expressed about military and civil requirements drifting apart.
The current situation calls for ambitious goals to be attained in commercial helicopters before the turn of the century. Designers of new military helicopters currently on the drawing board should be aware of these civil requirements and take them into account as much as possible.

1. INTRODUCTION

Numerous articles and lectures have been written about helicopter improvements. Some items are repeated over and over again, for example: safety, reliability, costs, comfort. They will be discussed again here, among other aspects that need to be highlighted as well.

Some people tend to be content with the current rate of progress. According to ref. 1 the helicopter is even approaching the point where it can be regarded as a routine method of public transport. Let me assure you that the helicopter is nowhere near that point:

- the newest medium-sized helicopter types are still unable to fully replace a 25-year old "workhorse" like the S61, essentially unchanged during its lifetime, because of its passenger comfort, its reliability and its low costs;
- helicopter noise still provides environmentalists sufficient reason to prevent heliports and services from being developed;
- all major helicopter operators still almost fully depend on the offshore energy industry, just like 20 years ago, because it is the only sizeable civil market that can afford helicopters and has no proper alternative available.

So let us not kid ourselves and determine where our goals should be and what can be done to attain them

A commercial helicopter operation is much different from a fixed wing airline operation. An offshore helicopter is operated from a local base to which it returns every day. Figure 1 shows the Dutch Continental Shelf with all its oil and gas platforms. The daily work of a helicopter involves short flights with many landings, that usually amount up to twenty, sometimes forty, a day. The average number of take-offs is 2.7 per flight hour for the S61 and 3.5 per flight hour for the S76; the difference mainly stems from a difference in cruise speeds.

Figure 2 lists the North Sea operators and the equipment they use. None of them is big enough to support a vast Research & Engineering Department. This may present a problem as the helicopter is steadily getting more complex.

A commercial helicopter operation is also much different from a military helicopter operation. It therefore sets different design requirements for example with regard to payload versus range, cabin size and seating arrangements, and ditching requirements. Some military requirements do not apply to civil operation at all, which in some cases renders a military helicopter useless for civil transport. Such cases are not limited to attack helicopters only. Since a manufacturer of a military helicopter will always start looking for civil customers for the same machine at one point in time it is useful to compare the military with the civil requirements at an early stage of the design process. The EH101 is a good example of this practice.

2. TYPICAL COMMERCIAL OFFSHORE REQUIREMENTS

2.1. PAYLOAD RANGE PERFORMANCE

The most obvious predecessor of an offshore helicopter is a military utility helicopter. Payload-range requirements however may differ widely. The UH-60 Black Hawk doesn't have enough range for offshore operations.
The EH101 on the other hand has plenty of range and plenty of payload because of its ASW-requirement. It might even be more than the offshore operators will need or be able to pay for. Time will tell.
Figure 3 shows how none of the above is the ideal S61N successor.

2.2. CABIN SIZE AND SEATING

Among typical military requirements such as agility, maneuverability, detectability, vulnerability, weapon system compatability and survivability there is air transportability. This is what caused the UH-60's telescopic rotor mast and low cabin ceiling. Its cabin comfort therefore is not able to compete with an S61 or Super Puma, both with aisle, as figure 4 clearly shows. Offshore helicopters are often used for shuttling, which means that a helicopter goes to many rigs in one flight, transporting passengers to and from each rig. An aisle in the cabin is a very valuable asset for this type of operation.
In addition, the baggage handling and refueling must be easy in order to limit ground times. Our goal should be airline standard, as in the EH101. The current S70C, the "Civil Black Hawk" has almost no baggage space. These examples clearly show that the S70C's fuselage would have to be modified considerably for it to become a commercially acceptable aircraft.

2.3. DITCHING

No offshore helicopter flying today, the S-61 being the only exception, was initially designed according to ditching requirements. Proper floatation gear and proper liferaft-storage have been a problem in every such aircraft. Even helicopters intentionally designed for civil use provide poor solutions, if any. The "solution" for the S76, for example, was to put it on a passenger seat. It took 6 years to solve it (figure 5); reasons: lack of manpower, funds and creativity.

2.4. MEDEVAC

The civil helicopter is increasingly being used for medical emergencies. Offshore operators have offered this kind of service already from the start, but their helicopters have not been equipped like today's dedicated Medevac aircraft. New designs should take this type of operation into account.

3. MAJOR IMPROVEMENTS ARE NEEDED

3.1. SAFETY

The British HARP report (ref. 2), mandatory reading for every designer, offers a clear view about current helicopter safety levels as compared to fixed wing aircraft. Figure 6 shows that when compared on a 'per hour of flight' basis the helicopters are worse than fixed wing airplanes. When treated on a 'per flight' basis helicopter rates would become comparable to propellor turbine transports but still fall short of jet transports. Statistics also show that military experience with a helicopter before it enters civil operation is beneficial to its civil accident record.

Figure 7 shows where improvements have to take place in order to improve the helicopter's safety record. Priority should be given to pilot aspects, such as cockpit environment, engine and airframe design and maintenance aspects.

3.2. RELIABILITY

Safety and reliability are highly correlated in helicopters. Technical delays and reliability are also highly correlated.
Technical failures delay 5% of all departures.
Figure 8 shows the total number of hours of delay during March 1986, caused by technical reasons, weather or other reasons.
A significant amount of the unreliability and failure of helicopters can be attributed to fatigue. The "high frequency" fatigue is the more significant case for helicopters, the source of loading being the rotation of the rotors.

3.3. AVAILABILITY

Availability numbers of helicopters typically range from 75% to 95% (90% means that the aircraft is available for commercial flying or training 329 days a year). Figure 9 shows the importance of reliable engines and quick replacement, vibration control, long TBO's and easy inspection procedures. Availability should of course be in the nineties (S61).

3.4. COSTS

Purchase prices of helicopters are much too high. Manufacturers claim that this is caused by the lack of sufficient and continuous production rates, high engineering costs, expensive materials, extensive testing, customer options, certification efforts, etc. and this is probably all true. Still, something should be done about it, because in the end commercial operators will only be able to buy used helicopters and production levels will remain low. A major goal for manufacturers should be to lower operating costs experienced by the customers. Apart from the obvious costs, the hidden costs represent a major unbudgeted expense (ref. 3):

OBVIOUS COSTS	HIDDEN COSTS
salaries	accidents
facilities	incidents
equipment	incompetency
maintenance	customer demands
training	management demands
fuel	
insurance	

The manufacturer's duty is to design and build a cheap helicopter that is as easy to maintain as a car, while it is the operator's duty to buy hundreds of them and operate them cheaply for everyone and everywhere.

3.5. CREW FATIGUE AND HEALTH

According to figure 7 32% of helicopter accidents/incidents are caused by pilot error. Contributing factors surely are pilot workload and crew fatigue. Crew fatigue can be caused by long duty hours, helicopter vibration, posture, seat (inability to move), continuous noise, many landings and take-offs.
These areas require specific attention.

Pilot health, back problems in particular, is also becoming a focal point of attention. Last year's record averaged 17 days of sick leave per pilot mainly due to serious long-term back complaints (normal population average should be 5 days). 80% of the pilots have back complaints (normal population average: 40%).
The main contributing factors are:

- asymmetrical posture of the spine due to specific helicopter controls (collective and stick); this is an inherent cockpit design problem;
- poor pilot seat configuration and adjustability;
- high vibration level

There are many reasons why this problem only surfaced fairly recent during the last five years. Examples:

- commercial pilots now average around or over 40 years of age and as they get older they become more sensitive to the problem;
- they average 14 continuous years of helicopter flying, logging thousands of hours, building up the problem;
- increased efficiency of duty hours with intense flying without enough intermissions.
  Military pilots might not have back problems to the same extent as commercial pilots, because of the reasons mentioned above. Nevertheless, every hour they log during their military career increases the chances of back problems that will surface during their commercial career later on.

  These problems should be a concern to the manufacturers also, but they seem to neglect it.

3.6. PASSENGER COMFORT

Seating, vibration and internal noise are the three main factors that contribute to the (lack of) comfort offered to today's helicopter passenger. Improvements should be possible, even in current helicopter types.

KLM's vibration limit is 0.2 ips. The industry's goal should be much lower: 0.05 ips for all frequencies. The most frustrating aspect of fighting vibration is the helicopter industry's basic lack of knowledge about vibration and rotor behaviour.
All manufacturers have their own experimental devices of decreasing or absorbing vibration, for example:

Sikorsky	-	bifilars, VTA and lateral vibration absorber
Aerospatiale	-	flexible gearbox attachment devices
Westland	-	Westland vibration absorber
Bell	-	noodle beam

Why don't they sit together, figure out one or two solutions and put that into all helicopters?
Why can't we accurately predict the effect of pitch rod and tab adjustments on the tracking picture of all blades, in forward speed conditions?
Why does MBB allow operators to adjust all tabs, while Aerospatiale does not allow any tab adjustments and Sikorsky allows only outboard tab adjustments (and what will Westland allow for the EH101)?
Why do manufacturers still think that calibrating a rotor blade on a two-bladed rotor tower (without forward speed) assures proper behaviour of that blade on a four-bladed helicopter in 140 kts forward speed?
Modern technology has given us swept rotorblade tips, but not the knowledge about their influence on vibration.

We don't see enough interest and real effort by the manufacturers, and operators also, to come to grips with vibration in order to substantially lower overall vibration levels. Let us hope that active vibration control will finally rid us of the problem.

3.7. EXTERNAL NOISE

Figure 10 shows how serious the problem is. We have made no progress in terms of external noise since the introduction of the S61 25 years ago. Quite to the contrary: more recent helicopters tend to get noisier. It is essential to the civil helicopter's future that this trend is reversed. External noise should become part of certification and credible and comparable noise measurements should be required. If that doesn't help, what will? We should not accept current noise levels as normal standard.

3.8. ALL WEATHER CAPABILITIES

In March and April of 1986 approximately 10% of all departures were delayed because of weather conditions.
Figure 8 shows the delays incurred by low visibility (or icing) conditions, many times leading to loss of revenues. Improvements should be made in these areas also.

4. WAYS TO IMPROVE

4.1. HEALTH AND USAGE MONITORING

According to ref. 4 the principal difficulties in implementing safe-life procedures are:

- predicting operating flight spectra and schedules;
- translating flight spectra into manoeuvres;
- determining loads produced in components by these manoeuvres.

Ref. 2 already noticed the wide disparity between the assumed operational duty used by the helicopter makers and what in practice operators achieve in practical operations. Manufacturers should specify more realistic operational spectra in conjuction with customers for their helicopters. A recent study performed inhouse by KLM Helikopters tends to support these views (figure 11); note how computer models often provide a false image of accuracy. Figure 11 also shows the extensive ground taxiing that is required on civil airports. This can cause fatigue damage, but no-one knows how much.

The only proper solution is usage monitoring, which should provide a more reasonable basis for retirement than flying hours. It should be set up in a way that includes onboard data reduction. This permits instant after flight status indication of each module without the need for further processing of data on the ground. The main attention areas are: engine trends, vibration and gearbox loads.

4.2. COCKPIT ERGONOMICS

A study performed by a KLM doctor and a KLM ergonomicist (human factors specialist) indicated clear directions towards solving pilot back problems:

1. a radically different cockpit design that allows a symmetrical posture of the pilot;

2. a pilot seat with the same features that every truck driver's seat has (and probably your own car as well): lumbar support, armrests, variable inclination, thigh support;

3. lower vibration levels.

This study supports the view of the HARP report (ref. 2) that human error-caused accidents may well be common to fixed wing aircraft pilots, and require ergonomic or psychological study as well as technical.

The only way to radically change the cockpit design is to incorporate side-stick controllers, that integrate the collective with the control stick. Ref. 5 shows that this is very well possible and sums up the advantages:

- increase in available cockpit space;
- up to 30% weight savings;
- improved reliability;
- pilot safety and comfort;
- improved visibility;
- better ingress/egress;
- improved crashworthiness;
- elimination of the poor posture caused by conventional controls.

Every newly designed helicopter should have side-stick controls. It is a grave pity that the EH101 doesn't have them. It would have been a major step forwards.

4.3. ROTOR BLADES

Improvements have been made in terms of aerodynamic design (performance), damage tolerance and fatigue (composites), but we got more complex vibration problems (blade tips, exchangeability of blades) and perhaps higher noise in return.
The first successful mixture of aerodynamic design and composite material seems to be the BERP-blade, currently flying on Westland Lynx and W30-300 aircraft and to be fitted on the EH101.
In fact, their improved aerodynamic design could only be achieved by the use of composite materials. If all the goals, including lower noise and vibration levels, are indeed attained by these blades, BERP blades should be on all helicopters within 5 years.

4.4. VIBRATION

The world according to HARP: the present high levels of vibration have a more or less serious effect on instruments and equipment on board helicopters and are a major source of fatigue damage to structures and components.
Eliminating vibrations means reduced fatigue failures and improved crew and passenger comfort.
Ways to improve:

- exchange of available knowledge among manufacturers and operators;

- vibration monitoring;

- better devices (the Westland vibration absorber for example should replace Sikorsky's bifilars if it's better and saves weight);

- better blade adjustment procedures.

4.5. AVIONICS

KLM Helikopters is content with current EFIS developments in terms of display format, redundancy, reliability and maintenance. Such improvements however do not outweigh a 70 lbs weight disadvantage as compared to conventional electro-mechanical instruments. As soon as the next generation arrives, with flat plate image and no additional coolers required, we intend to replace conventional instruments.

4.6. FLIGHT DATA AND COCKPIT VOICE RECORDERS

Aside from health and usage monitoring systems Flight Data Recorders should receive more attention.
Why are current recorders so heavy?
They should be light and low-cost in order to enable operators to install one in every helicopter. The additional facts about accidents acquired in such a manner will contribute immensely to a better accident rate and prevent a great deal of money spent on investigations. There are numerous examples, some of them quite recent, to support these views.

4.7. DESIGN PHILOSOPHY

Designing a new helicopter is not what it used to be.
Requirements are tougher. Systems are more complex.
The manufacturer's organisation is more complex and restrictive, especially when the design is a joint effort with another company. New helicopters are not built every 3 years anymore but every 10 years, which doesn't leave much room to build up experience.
A designer nowadays has to be cost-conscious; he also has to know what damage tolerance is and which kind of new materials are available. His main friend is a mainframe, that enables him to calculate complex loads and accurate dimensions, but doesn't tell him anything about ease of inspection or the dangers of Murphy's law. And the operator's hangar seems farther away from the designer's office than ever before. Is it really a surprise that 29% of accidents from airworthiness causes were due to "small but significant parts"? Technology is advancing, yet for all intents and purposes, man remains constant. Modern equipment not only often exceeds the capabilities of the military or civil user, it may also exceed the manufacturer's own design capability. Engineering judgement sometimes seems to be going out of style, in favour of "computerized solutions"! The elegance of simple solutions is often overlooked.

If designers would themselves have to operate and maintain their own helicopters we would surely see much improvement. They should make much more field trips to operators, while field representatives and operators should be allowed a much bigger input in the early stages of the design process. A special US Army program, called MANPRINT (MANpower-PerRsonnel-INTegration), ref. 6, is based on a new design philosophy.
"It makes no sense to design a system to operate at a peak or plateau that cannot be operated or supported by the human", whereas the past philosophy was to design the equipment/systems and then mold the operator/maintainers to fit this design.
This type of design philosophy has in my opinion greatly contributed to the success of automobiles. If it works there, why shouldn't it work for helicopters?

4.8. DEVELOPMENT TESTING

HARP says: "not all residual problems or defects will have been eliminated when a helicopter first enters service. The proper procedure has been and should be for the manufacturers and the individual operators to cooperate fully to eliminate early defects. Monitoring "fleet leading" machines is an increasingly general practice.
More accurately: the helicopter operator is actually performing the certification himself; after 4 years in commercial operation you know where the real problems are. Why then is he not equipped as a flight test department should be: with flight data recorders from the very first day onwards; with pilots that are instructed to behave and respond as test pilots; with engineering staff instructed to interpret flight data and handle flight data reductions? There is much room for improvement in the area of commercial development testing. Lead-the-fleet practices (sending back components for detailed inspection) is just part of the solution.

5. CONCLUDING REMARKS

1. The authors of ref. 7 voice a typical manufacturer's point of view as they conclude that today's performance, comfort, reliability and predictability are of such order that was only dreamed of in 1943. Lowering purchase prices and operating costs however has not been possible.
Today, in 1986, civil operators still find themselves dreaming; dreaming of product improvements that are absolutely essential for the development of the commercial helicopter industry.
If helicopters are to extend their operations into areas where they can compete with fixed wing aircraft, and, even more so, local ground transportation (taxi, bus and train) their inherent safety and passenger facilities and comfort, noise and vibration levels must be improved, as well as their costs.
Why doesn't every small city have its own busy heliport like it has a railway station? Because the right helicopter is not available yet.
My question is: can manufacturers make it available before the turn of the century? Figure 12 lists the type of improvement that is needed for the current and next generations of commercial helicopters. Perhaps a new design revolution is required. If it doesn't happen the civil helicopter industry is destined to rise with the offshore energy industry and to go down with it, as it is doing now in a 7 year-cycle. If you hadn't noticed: the trend is downward at the moment.

2. The last Flight Mechanics Panel Symposium devoted to rotorcraft design was held in 1977 at NASA-Ames.
The technical evaluation for that meeting (AGARD-AR-114) expressed concern that military and civil helicopter developments are drifting apart in design features and specific recommendations. This is exactly right, maybe in 1986 this concern should even be greater than in 1977. We have seen the entry of a splendid utility Black Hawk with high production levels (low purchase price) and many new features rendered useless to the commercial world due to specific military requirements. We are seeing a splendid new EH101 with many desired improvements, that may very well be too big and too expensive for commercial use due to its shear size due to military (ASW) requirements.

Some people might conclude that the two users are not so far apart as was feared, because some civil versions of military helicopters and military versions of civil helicopters have entered the market since 1977. Such a conclusion is wrong: the manufacturers and operators simply have no choice and there are no successes to be recorded. We can only hope that in future design efforts manufacturers will bring together basic military and civil requirements at early stages of the design process, as EH Industries has been trying to do with the EH101.

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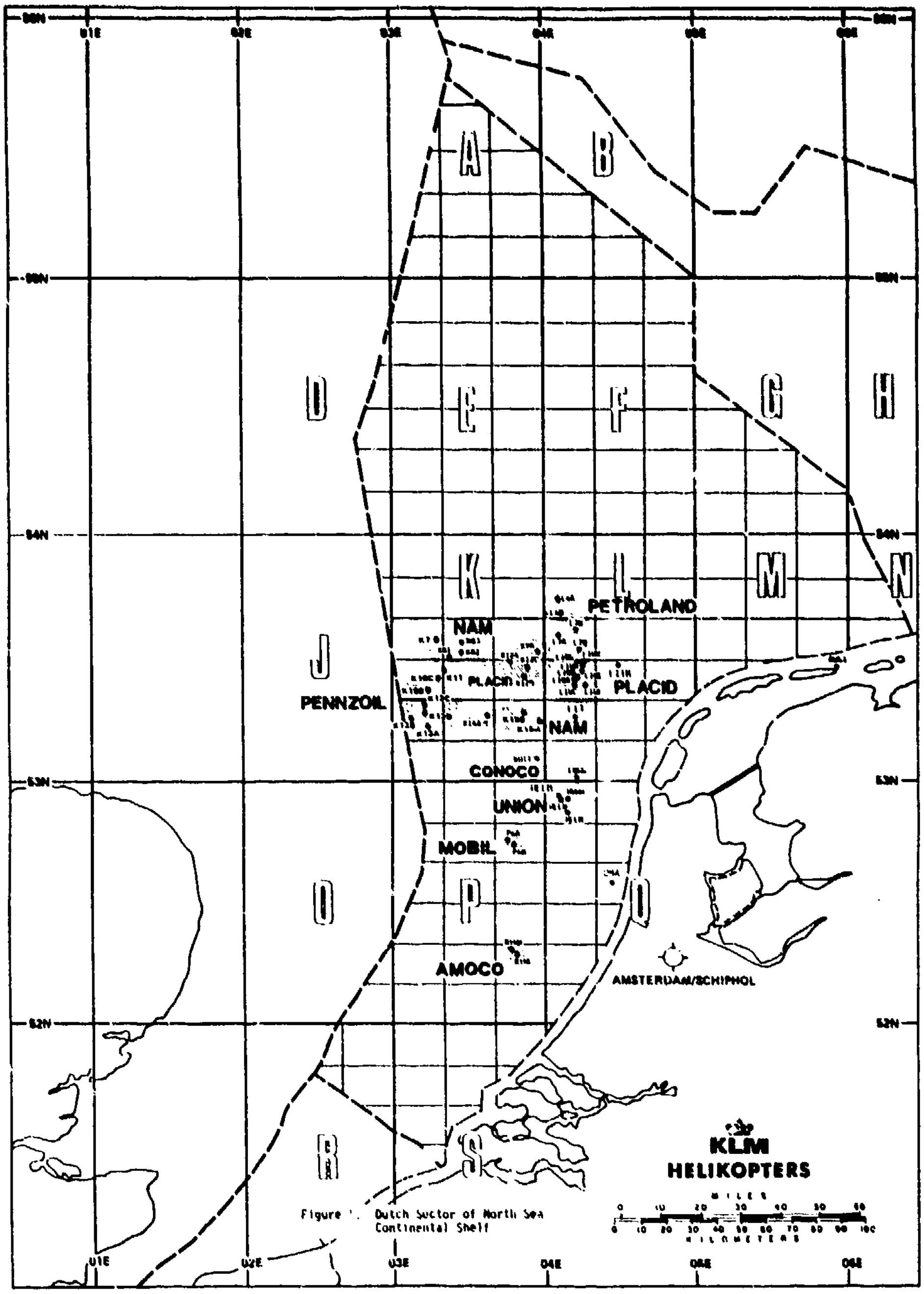

Figure 1. Dutch Sector of North Sea Continental Shelf

Figure 2 *NW EUROPE CONTINENTAL SHELF - SUPPORT HELICOPTER FLEETS (Source: Heli Data sept. '86)

	BAH	BCHL	BHL	BOND	HEL SVC	LFT TPT	MSK	KLM	WKG	IRH	TOTAL
CHINOOK	4	–	–	–	3	–	–	–	–	–	7
AS332L	3	–	19[2]	7	7	6	2	–	–	–	44
S-61N	11[2]	3[2]	11[2]	–	19[3]	–	–	8	–	2	54
214ST	–	3[2]	–	–	1[1]	–	–	–	–	–	4
212	–	–	7[3,4]	–	8[4]	3[5]	3	–	2[6]	1	24
S-76A/B	3[2]	1	6[3,4]	5	–	–	–	4[6]	2[5]	–	21
SA365N	–	–	–	8[4]	–	–	1[1]	–	–	–	9
SA365C	–	–	–	5[2,5]	–	–	–	–	–	–	6
W30	3[1]	–	–	–	–	–	–	–	–	–	3
Bo105D	–	–	–	7[3,4,6]	–	–	–	–	–	1[5]	8

Notes: 1 = Includes leased aircraft
2 = Excludes aircraft withdrawn from use/for sale/leased abroad/on non support tasks
3 = Exact operational holding uncertain
4 = Some based offshore
5 = All not engaged on support tasks
6 = Being replaced by S-76B 1986/7

(BAH - British Airways Helicopters; BCHL - British Caledonian Helicopters; BHL - Bristow Helicopters;
HEL SVC - Helicopter Service; LFT TPT - Lufttransport; MSK - Maersk; WKG - Wiking; IRH - Irish Helicopters)

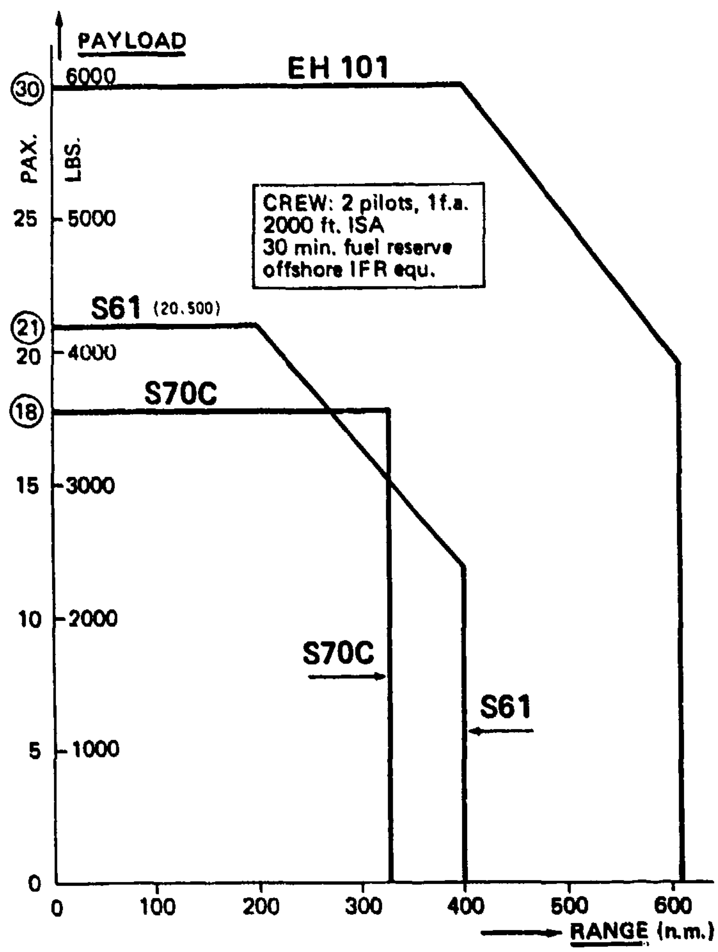

Figure 3. Payload - Range comparision S61, S70C, EH101

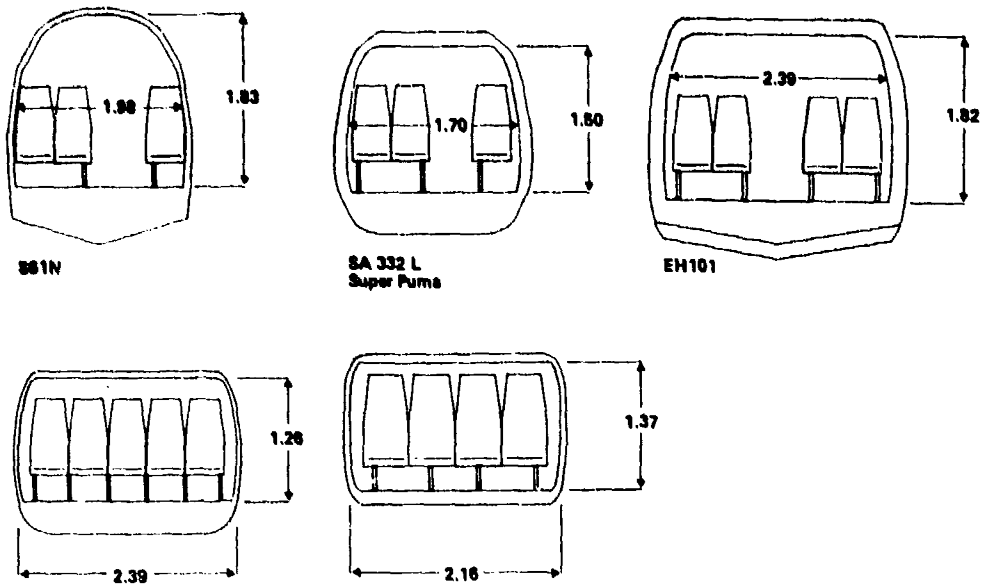

Figure 4. Comparison of seating arrangements, medium-- sized helicopters

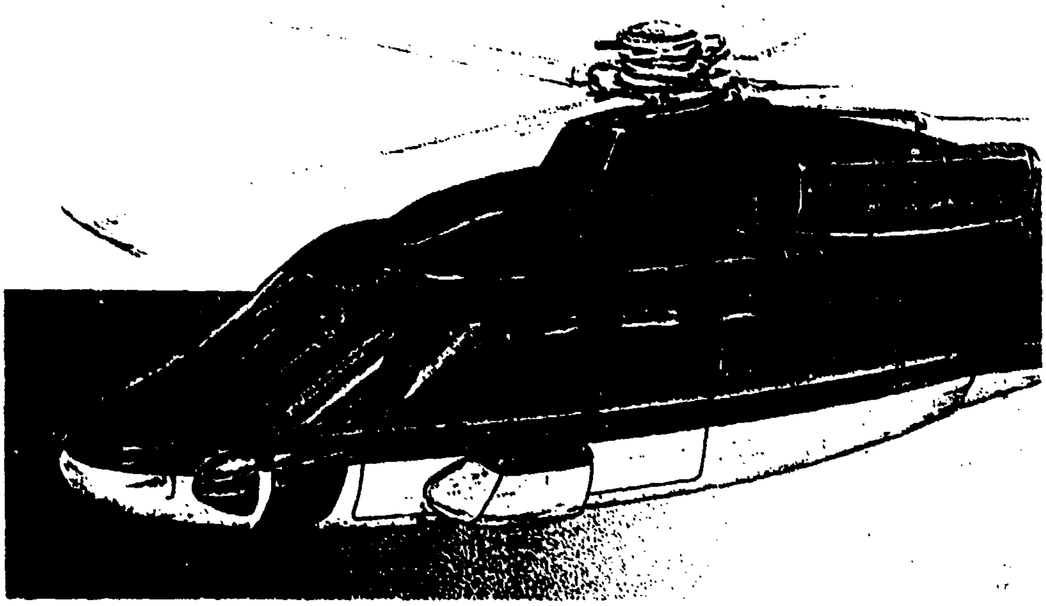

Figure 5. New life raft location S76-B

	Jet Transport (100 seat)	Prop Turbine Transport	Typical Helicopter
All accidents	0.4	2.0	2.0
All airworthiness accidents	0.08	0.5	1.0
Fatal airworthiness accidents	0.01	0.07	0.5

Figure 6 Accident rates per 100.000 hours
(Source: ref. 2)

Pilot	Engine	Airframe	Maint.	Misc.	Weather	Total
32%	29%	16%	13%	6%	4%	100%

Figure 7 Typical accident/incident cause analysis
(Source: ref. 3)

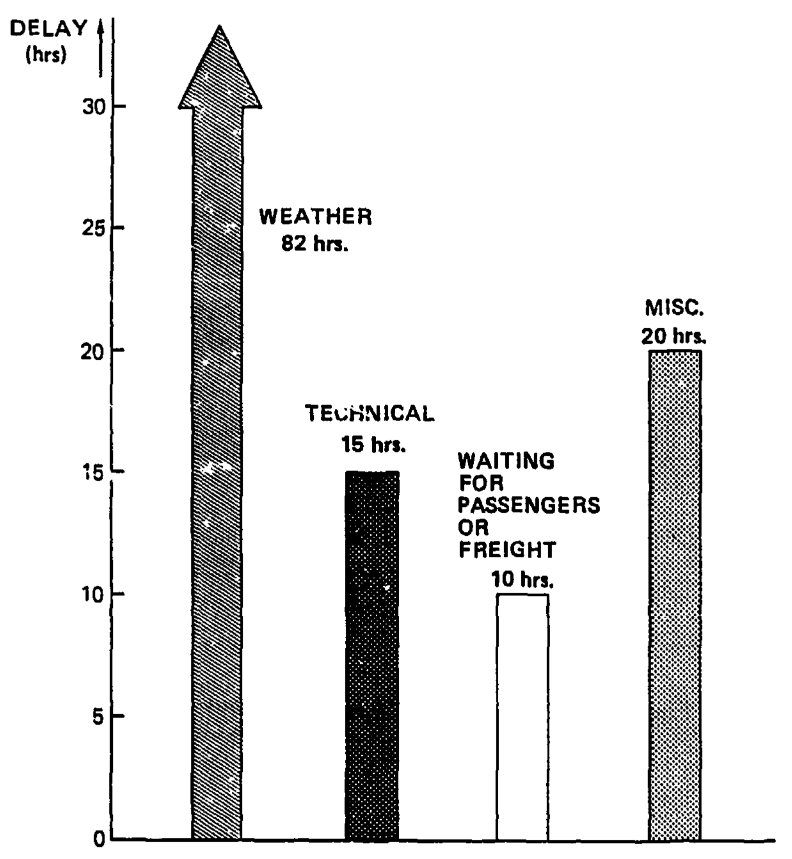

Figure 8. Total delays by weather, technical and other reasons.
(KLMH Schiphol - Operation, March 1986, 385 departures)

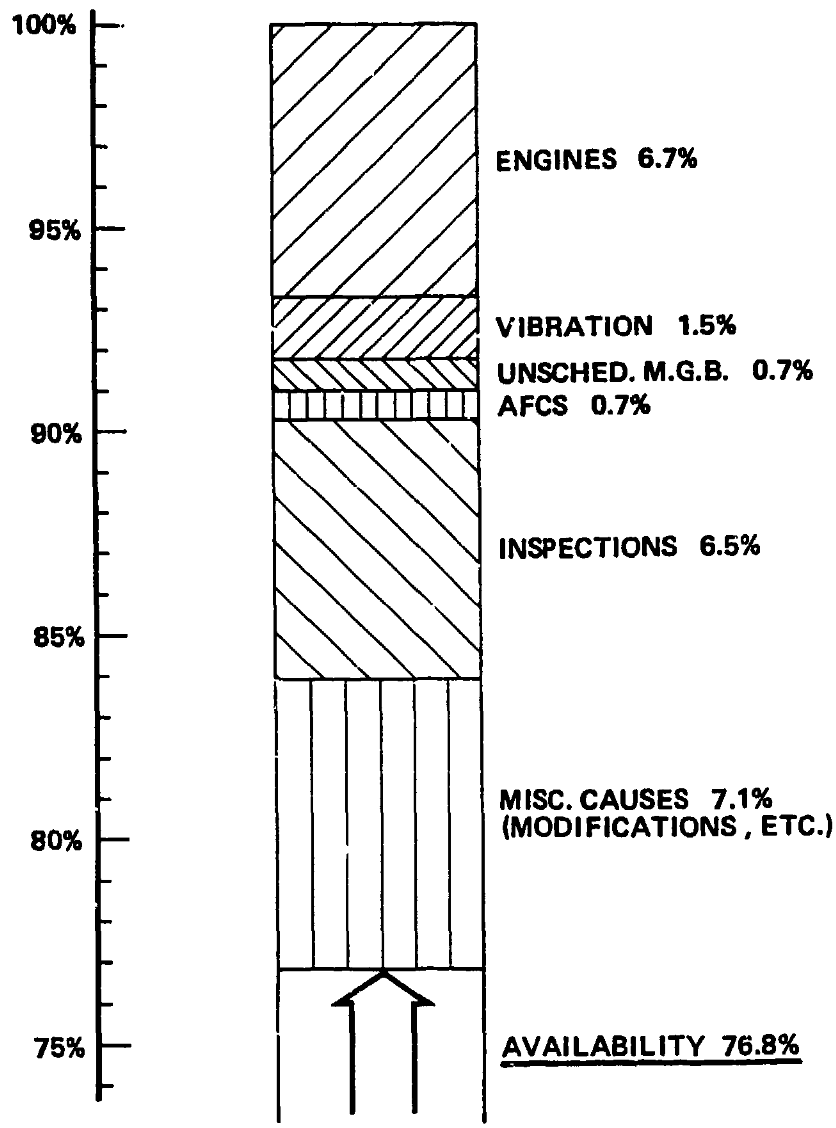

Figure 9. Example of technical availability losses

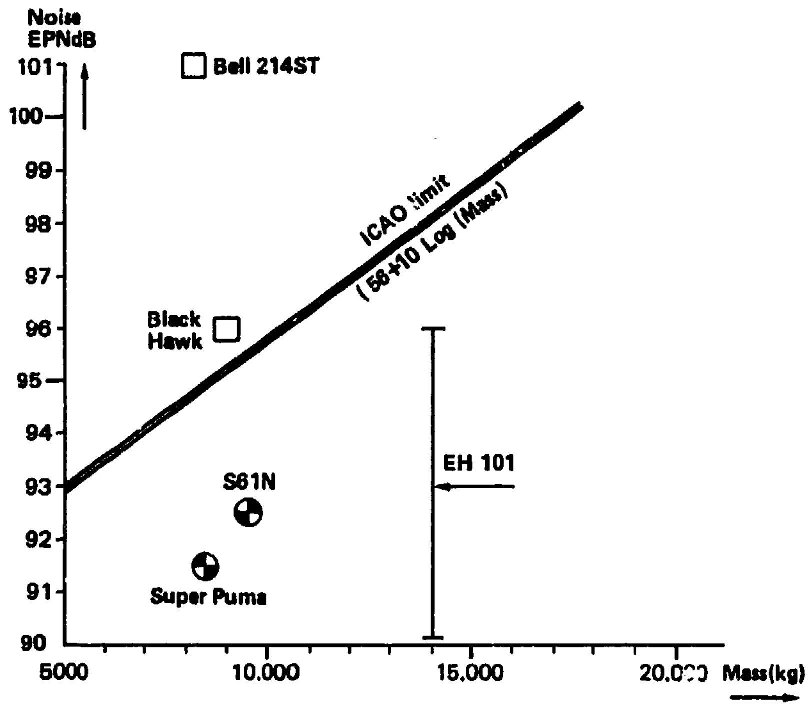

Figure 10. Fly-over noise of medium-sized helicopters (alt. 150m, Vmax level flight)

flight condition	percentages per flight hour		
	Computer model	KLM actual data	
	Seaking Transport	S61	S76
hover	2.83%	1.3%	1.4%
take-off: rotation	?	1.2	1.6
acceleration	?	0.8	1.0
climb	0.57	13.5	17.5
level flight	91.48	61.8	51.1
left turns	0.57	2.6	3.2
right turns	0.57	2.6	3.2
autorotation	0.62	+ 0	0.1
descent	0.34	13.5	17.5
landing : deceleration	?	2.3	2.9
approach to hover	?	0.4	0.5
total inflight	96.98%	100 %	100 %
taxi	?	7.6	9.9
engine running	?	11.2	13.3
total active time		118.8%	123.2%

Figure 11 Comparison actual flight spectra of KLM S61 and S76 with research computer model

MGW 8000 - 15000 lbs	MGW 15000 - 32000 lbs	IMPROVEMENT GOALS
FIRST GENERATION (entry '60 - '75)		
S58T Bell 212	S61N AS330 Puma	not much
SECOND GENERATION (entry '75 - '90)		
Bell 222 SA365C/N S76A, B, etc. Bell 412SP Westland 30	Bell 214 ST AS332L Super Puma (S70C)	- better/uprated engines - increased number of blades (Bell: 2 to 4; Westland: 4 to 5) - composite rotor head - composite Fenestron - BERP blades - Westland vibration absorber and other devices - EFIS without weight penalty - first step health monitoring - increased TBO's/life limits - engine FADEC - 4-axis autopilot systems with proper Flight Director - pilot seat - de-icing - MLS
THIRD GENERATION (entry beyond '90)		all of the above plus:
civil utility A129? LHX civil spin off?	EH101 modified S70C? Super Super Puma? civil NH90? civil tilt rotor?	- better payload-range - composite fuselage - better OEI performance - lower purchase price (50% of second generation) - one or less maintenance manhours per flight hour - fixed wing safety level - 98% availability - 0.05 ips vibration - 80 db internal noise - 85 EPNdB fly-over noise (less for smaller helicopters) - 150 kts normal cruise speed

FIGURE 12 THREE GENERATIONS OF OFFSHORE HELICOPTERS 8000 - 32000 lbs MGW

MISSION-ORIENTED FLYING QUALITIES CRITERIA FOR HELICOPTER DESIGN VIA IN-FLIGHT SIMULATION

by

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SUMMARY

The increasing demands on and the complexity of future highly augmented rotorcraft call for extensive simulation activities in the field of flying qualities utilizing operational helicopters, ground-based, and in-flight simulators. If both, variable system characteristics and high realism of pilot/helicopter situations are required in-flight simulators demonstrate definite advantages.

The key contributions of in-flight simulators in the fields of rotorcraft system and subsystem design and for establishment of handling qualities criteria are discussed on the basis of DFVLR experience and activities. By presentation and discussion of recent results achieved with the in-flight simulator ATTHeS, operated by DFVLR, the potential of this research vehicle in particular and of advanced rotorcraft in-flight simulators in general is demonstrated with respect to design and development of future helicopter systems.

1. INTRODUCTION

New missions for civil and military rotorcraft lead to increasingly higher requirements in the field of flying qualities. These demands on and the complexity of future rotorcraft call for extensive simulation activities during the execution of development and research programs.

The simulation methods and facilities available for flying qualities investigations include

(1) theoretical studies and analytical computer simulations using large scale computers in non real-time applications,

(2) experimental research and system development with scaled models in wind tunnels,

(3) real-time ground based simulation with the pilot in the loop,

(4) airborne simulation using variable stability and control aircraft in operational environment, and

(5) flight testing with prototypes or operational rotorcraft.

None of these research and development tools can be omitted because of the different tasks and objectives of the programs as well as the inherent advantages and disadvantages of the specific methods and facilities (Refs. 1, 2). For in-flight simulation a unique role complementary to all other simulation methods is seen and will be demonstrated in this paper in the light of relevant DFVLR experience and activities.

During a development program objective procedures are required to validate the design parameters as early as possible and to answer the question wether the rotorcraft system design meets all flying qualities specifications. In order to fulfill these needs of system evaluation most realistically, in-flight simulation is the ultimate assessment method providing high realism and credibility but much lower cost and complexity than prototype flight testing, as shown in Figure 1. The main objectives of in-flight simulator utilization in this respect are

(1) reduced development costs and risks,

(2) control software validation, and

(3) enhanced pilot-in-the-loop acceptance testing.

Complementary to Figure 1, the role of in-flight simulation for the establishment of handling qualities criteria is demonstrated in Figure 2. In such a research program high realism is also required, of course, but in addition the flexibility is of decisive consequence. In-flight simulation offers much higher flexibility than operational rotorcraft or prototype flight testing, allowing the generation of a generic flying qualities data base covering a broad spectrum of dynamic rotorcraft characteristics (Ref. 3).

The benefits of in-flight simulators over ground-based simulators include perfect motion and visual perception and the realistic impact of truly flying a helicopter. Experience has shown that lack of complete motion and visual cues in ground-based simulations sometimes tends to limit rapid control actions necessary for high-workload manoeuvering (Refs. 4, 5).

In this paper the utilization of DFVLR's helicopter in-flight simulator ATTHeS (Advanced Technology Testing Helicopter System) for rotorcraft system and subsystem design as well as for the establishment of rotorcraft flying qualities criteria will be presented, demonstrating the high potential of advanced in-flight simulators for design and development of future helicopter systems.

2. IN-FLIGHT SIMULATOR ATTHeS

Since 1982 the research helicopter BO 105-S3 is operated at DFVLR-Braunschweig (Figure 3). The helicopter provides a fly-by-wire flight control system and was recently completed to become the in-flight simulator ATTHeS.

Basic System

The basic research helicopter BO 105-S3 corresponds, in all essential components to the serial helicopter MBB BO 105 with the exception of the control system. The modified system requires a two-man crew consisting of a safety and an evaluation pilot for simulation flights. The safety pilot, who occupies the left-hand back seat, is provided with a direct link to the primary helicopter controls through the standard mechanical/hydraulic control system. The evaluation pilot, seated in the center in front of the cockpit, has either conventional or side-stick controllers which are electrically linked to the helicopter controls. A simplified schematic diagram of the control system is shown in Figure 4. The fly-by-wire system is a full-authority, simplex system (Ref. 6).

The helicopter can be flown in three modes: (1) the fly-by-wire disengaged mode, where the safety pilot has exclusive control, (2) the fly-by-wire 1:1 mode, where the evaluation pilot drives the control system, and (3) the fly-by-wire simulation mode, where additional control signals are generated by an on-board computer realizing specific control laws for the simulation pilot.

When the simulation pilot station is engaged, the actuators operate in an electrohydraulic mode with mechanical feedback to the safety pilot's controllers. The safety pilot can override the fly-by-wire system by applying a specific force to the appropriate controller. In addition, the system can be disengaged by both, the simulation pilot or the safety pilot and by an automatic safety system using pre-set limitations in selected sensor signals.

Model-Following Control System (MFCS)

For realization of variable stability and control characteristics a model-following control system for the research helicopter BO 105-S3 was designed (Ref. 7). This helicopter, using a hingeless rotor system, is especially suitable for an in-flight simulator with respect to the high control power available. However, the inherent interaxis couplings, the strong nonlinearities and existing uncertainties in the system parameters require a level of performance from the variable stability system that is difficult to be achieved.

In general, control of the dynamic response characteristics is accomplished using either response feedback and feedforward techniques, or model-following control systems. The comparison of both techniques demonstrates essential advantages for the MFCS, particularly due to the capability to suppress real turbulence including wind-shears without maneuver response (Ref. 8). In addition, an MFCS is very flexible and allows quick and easy adaptation to helicopters with different dynamic characteristics.

In a typical MFCS the pilot's commands are disconnected from the actual aircraft and fed into a model. This model represents the dynamic equations of motion of the aircraft to be simulated. The errors between the response of the model and those of the host aircraft are fed into the control system, which attempts to minimize the state errors by generating control signals for the actuators. If the state errors are always zero, the controlled vehicle exhibits the dynamic characteristics of the model.

The structure of the MFCS is shown in Figure 5. In addition to the feedback control loop, a feedforward control loop is installed to accelerate the host helicopter response and to reduce the necessary feedback control gains.

In order to evaluate the performance and limitations of the MFCS and to qualify the system for use in helicopter flight research, a U.S./German simulation experiment was conducted on a ground-based simulator at NASA Ames Research Center. This joint research program was part of the U.S. Army/DFVLR activities under the U.S./German Memorandum of Understanding on Helicopter Flight Control.

MFCS-Realization

In order to provide the computer capacity necessary for completion of the different functions, the MFCS is realized with a multi-computer system using four PDP-type computers, as shown in Figure 6 (Ref. 9).

The mathematical model to be followed and the control laws of the MFCS are implemented in the **Data Control Computer**. In addition, this main computer accomplishes the communication between all computers, the handling of the sensor inputs and the controlled outputs to the fly-by-wire actuators, the calibration of the sensor signals using off-line data, and the transformation to engineering units. For signals, which cannot be measured directly, the main computer incorporates transformations and state observers. The Data Control Computer is connected to a mini-tape recorder for reading specific flight test parameters and recording system parameter changes.

The sensor data are processed by the Data Acquisition System providing programmable gains, offsets and cut-off frequencies of six-pole Bessel filters. Then these data are used as input for the Data Control Computer as well as for PCM data transfer to a mobile ground station providing quick-look analysis for the flight test engineer.

The communication between the simulation pilot and the on-board computer system is accomplished by the **Pilot Communication Computer**. By operating a numerical keyboard in combination with a cockpit display using menu-techniques the simulation pilot is able to select and to initiate different functions of the control system.

The data from the Data Control Computer are transferred to the **Data Recording Computer** which is connected to a floppy-disk drive, where all helicopter and control system states are recorded during the flight test runs.

The **Ground Simulation Computer** enables a real-time ground simulation by simulating the actuators, the rigid-body dynamics of the helicopter and the Data Acquisition System of ATTHeS. The Ground Simulation Computer can be engaged simply by plugging a contact into the front panel allowing to test the total software for the in-flight simulation in a ground simulation run.

System Performance

To demonstrate the performance of ATTHeS and MFCS different existing helicopters were simulated in flight, including a SA 330 Puma helicopter as well as a UH-60 Black Hawk. Figure 7 shows time histories of the input signals and the response of the UH-60 model compared with the measured response of the in-flight simulator. The good curve fit between commanded states and real helicopter states demonstrates the capability of ATTHeS as an in-flight simulator.

Depending on the characteristics of the basic helicopter, of the equipment installed, and of the operational conditions the capabilities of in-flight simulators are limited, of course. The exact knowledge of these limits is assumed to be essential prerequisites for conducting successful flight test programs.

Due to different factors, e.g. model bandwidth, sample time of the control system etc., it is not possible to realize the exact dynamic model characteristics with the in-flight simulator. Therefore, quantitative analysis is required to estimate and evaluate the actual simulation performance with regard to the specific test progams.

Because only four controls are normally available in a helicopter, the motion can be independently controlled only in 4 degrees of freedom. The realization of a 6 DOF helicopter in-flight simulator requires additional longitudinal and lateral force-generating capabilities. The limitations arising from this point influence the longitudinal and lateral translational motions and, in addition, the turbulence response characteristics of the simulator.

Severe restrictions in the flight envelope may be raised by flight safety aspects. In general, it will be inadmissible and has to be avoided to operate extremely close to the ground using a simplex fly-by-wire control system only. The reasons for this are, of course, the lack of redundancy in case of failures in the system, but in addition, the impairment of the evaluation pilot's behavior by his knowledge of the safety critical situation. By means of dedicated safety pilot training programs flight safety can be improved substantially. This includes the simulation of critical system failures at safe altitudes and the subsequent taking over of the controls from the evaluation pilot by the safety pilot for helicopter state recovery. In this respect, the safety features incorporated in the in-flight simulator are of decisive consequence. Very essential devices used in the ATTHeS in-flight simulator are:

(1) an automatic safety trip that disengages the fly-by-wire system if a system failure occurs or if a sensor measurement exceeds a level corresponding to a structural or flight condition limit,

(2) a control monitoring system which aids the safety pilot in diagnosing a failure status. In addition, the safety pilot is involved in the flight test program very early in order to improve his test specific monitoring capabilities.

Up to now the ATTHeS in-flight simulator was used in different test programs, as shown in Figure 8, representing approximately 300 flight hours. These flight tests include the development and update of the systems as well as applications in handling qualities studies.

In order to further improve the in-flight simulation capability the realization of an operational in-flight simulator HESTOR on the basis of the helicopter BK 117 is

planned by DFVLR and MBB. The application of this system is mainly intended for research activities and for tasks in the area of development, testing, and integration of new technologies with regard to design, development and certification of future rotorcraft.

3. ATTHeS UTILISATION FOR ROTORCRAFT SYSTEM AND SUBSYSTEM DESIGN

The expansion of the role of rotorcraft constrains the application of new key technologies like

- digital electronics and avionics,
- active control systems, and
- related software technology

in the design of future rotorcraft in order to optimise the pilot-helicopter system with regard to

- system-performance and operation,
- design-facilitation and flexibility, and
- development and operational cost reduction.

In turn, the innovative potential of these key technologies are causing increasing concern with respect to the

- helicopter system complexity, and
- flight critical behavior with regard to extreme flight regimes and subsystem failures.

Therefore, a well balanced compromise has to be found between the specified mission performance, the pilot workload, and the costs, addressing all flight configurations under normal operational conditions as well as in failure situations.

The conceptional design of the overall helicopter system with respect to flying qualities has to take into consideration the characteristics of the individual subsystems, the integration of the subsystems and the influences on the overall helicopter system performance evaluation. These subsystems include the

- basic helicopter with its inherent stability and control characteristics,
- flight control augmentation systems,
- pilot information systems,
- controllers, and
- operational equipment.

Especially for [illegible] evaluation of the applicability and effectivity of new technologies for specific subsystems, their influences on the overall system, and their integration problems the in-flight simulation offers a unique potential for utilization during the design, development and certification process (Ref. 10). This capability will support industry to meet the demands for future rotorcraft systems and to reduce costs and risks of development programs.

Key contributions are expected in the following fields:

- basic design parameter investigation,
- pre-production design verification,
- flight control system development,
- hardware in-flight testing
- simulation of system [illegible] states,
- pre-first-flight pilot training, and
- support during certification procedure.

The capability of the ATTHeS in-flight simulator in this regard has been demonstrated in different research programme including the

- implementation of mathematical helicopter models representing a broad variety of essential design parameters (as shown before),
- realization of full-authority flight control systems and autopilots including the MFCS as a robust adaptive feedback controller, and

- realization of specific control-response systems.

As an example, Figure 9 shows the implementation of an attitude command (AC)-attitude hold (AH) response system for the roll axis including decoupling of roll and pitch motion. The frequency response as well as the time histories measured during flight tests demonstrate the accuracy of the in-flight simulation with respect to the response system model.

Additional response systems realized and tested with ATTHeS include turn coordination for the pedals, rate of climb response for the collective controller, rate command (RC), rate command (RC)-attitude hold (AH), and attitude command (AC) for the stick controller.

For future utilization of the in-flight simulation facility with respect to technology evaluation, the integration of a side-stick controller is under development, providing a broad variability in controller modes. In addition, a programmable pilot display system will be integrated. Further test programs will cover especially transients in failure situations as an essential aspect for application and certification of new technologies.

4. ATTHeS UTILIZATION FOR ROTORCRAFT HANDLING QUALITIES CRITERIA

The existing handling qualities criteria, especially for military applications, are no longer suitable as a design guideline and for the evaluation of new helicopter projects. Therefore, the U.S. Army initiated a program with the objective to develop new mission-oriented criteria, addressing the aspects of the expanded role of helicopters and the application of new technologies (Ref. 11).

Experience in previous efforts to revise the handling qualities criteria showed that the primary handicap to developing new requirements was the lack of systematic data from which new criteria could be defined and substantiated. These data gaps were recognized also during the ongoing attempt (Ref. 12). In order to expand the data base and to obtain generally valid results systematic tests are required using ground-based simulators, operational helicopters and especially variable stability helicopters. This data generation seems to be both, very important and voluminous requiring the cooperation of all institutions having relevant experimental capabilities (Ref. 13).

Key contributions to be made by in-flight simulators include the following.

- Control system/display relationship.

  Several studies have shown that for constant pilot workload a tradeoff exists between control system complexity and cockpit display sophistication. In other words, this hypothesis says that a very advanced pilot information system could compensate for a degraded flight control system and a very advanced flight control system would minimize the need for display sophistication. Together with the consideration of cost, which normally increases with complexity, these relationships seem to be very essential for the design of future helicopter systems. There are many and good arguments for the general validity of this hypothesis but, there are only a few mission-oriented flight test data available for the design engineer or for the development of generic requirements (Ref. 14).

- Control response characteristics.

  There is a lack of data for specification of control system response types necessary to guarantee Level 1 handling qualities for specific missions in different environmental conditions. For possible control system responses the required parameters like bandwidth, time delay, damping ratio, controller sensitivity and command gradient have to be defined for both, center- and side-stick controllers.

- Cross-coupling.

  Control and vehicle cross-coupling characteristics have a fundamental influence on pilot workload and task performance. The data available from ground-based simulation programs need to be verified using flight-test data. In addition, new data generated in realistic operational environment are necessary for specification of requirements.

- Degraded handling qualities.

  An important aspect of the development of new criteria, that was sometimes overlooked in the past, is the determination of boundaries defining minimum acceptable standards for civil criteria, or to meet Levels 2 and 3 of military flying quality specifications. Up to now there are only few data available to specify limitations for degraded handling qualities following failures in control systems and vision aids. In addition, the transient behavior concerning control system mode switching and failure situations is of decisive consequence for sophisticated control systems. These requirements seem to be very sensitive to the operational environment and the pilots' capabilities. Therefore, tests using in-flight simulators are required.

Because of these shortcomings one main objective of DFVLR activities is to establish a data base for the development of generic criteria and for the support of specific helicopter projects. During these activities operational helicopters have been utilized for the assessment of mission demands and for derivation of representative flight test tasks (Ref. 15). The ATTHeS in-flight simulator is primarily used for investigating the influence of specific parameters representing a broad variety of helicopter characteristics, control laws, and mission-dependent equipment.

A test program specifically planned to contribute in filling the data gap of the new helicopter military specifications was a study of roll response required in nap-of-the-earth slalom flight (Ref. 16). In these test the roll damping and roll sensitivity of the helicopter were varied in order to evaluate the dynamic roll characteristics with respect to the flight task. The pitch response was adapted in harmony to the roll behavior and the initial coupling response of the basic helicopter was changed to a level desired by the evaluation pilots. This adaptation of the off-axis response was achieved during a pre-test using ground-based simulation experience as well as pilot evaluations during flight tests.

In Figure 10 the evaluation results of this roll control study are summarized and compared with previous studies and criteria from the specification MIL-F-83300 (Ref. 17). These data were generated under different conditions and do not comply directly with the agressive low altitude slalom manoeuver of DFVLR and therefore the differences underline the task dependence of the results. The flying qualities level boundaries drawn by the DFVLR data require higher roll damping-control sensitivity combinations for satisfactory handling qualities (Level 1) in the low altitude slalom than MIL-F-83300. The tests suggest a control sensitivity ($L_{\delta y}$) of 1 to 2 rad/s^2/inch and a damping (L_p) of -4 to -8 s^{-1} for Level 1. The recommended Level 2 boundary (acceptable) is close together with the Level 2 requirement of MIL-F-83300. This result may reflect the fact, that flight test data with helicopters having inherent high roll sensitivity were not available at the time of MIL-F-83300 development.

During recent activities to update the specification MIL-H-8501 a concept for high amplitude roll manoeuver criteria based on the phase plane technique was proposed (Ref. 18). The required task performance in this concept is described by the commanded net roll angle changes and the corresponding peak roll rates during these manoeuvers. Figure 11 shows the data of DFVLR experiments in comparison to the proposed MIL-Spec criteria. These criteria are based on test data produced mainly with operational helicopters in hover tasks and adopted for forward flight. Areas of discrepancies between MIL-levels and levels drawn by DFVLR data can be recognized, showing that the MIL requirement especially for Level 1 is more stringent than the DFVLR results.

In addition, the DFVLR data underline the fact that the criteria have to be extended for higher bank angle changes. In this region the data show a saturation of peak roll rate. This saturation level is clearly dependent on the flight task, which has been demonstrated by recent additional tests performing so-called figure eight manoeuvers.

These results show, that handling qualities investigations having the objective to establish generic criteria require flight test data produced with test aircraft having adequate performance and covering a broad area of aircraft characteristics. Otherwise, specific phenomena will not be detected, yielding criteria which may be very restrictive with respect to the application of new technologies.

An additional aspect, supporting the statement that test aircraft need to have adequate performance and variable characteristics for handling qualities tests, is illustrated in Figure 12. In these tests side step manoeuvers in the low speed region were performed using a operational BO 105 with mechanical control. The BO 105 response in this condition can be approximated as a rate response system. The flight test data support very well the proposed Level 2 handling qualities but no Level 1 data could be achieved. There may be two reasons that the pilots evaluated the BO 105 helicopter with its quick roll response (as commented by the pilots) as Level 2 handling qualities: (1) the high level of inherent couplings yields high pilot workload during this task, (2) a rate response may not be adequate for high amplitude roll manoeuvers in hover.

Therefore, operational helicopters although adequate with respect to the required task performance may not be adequate with respect to the generation of handling qualities data for the establishment of generic criteria. These data can be produced only by test vehicles having in addition variable characteristics with regard to all relevant parameters.

Further utilization of ATTHeS for handling qualities evaluations is planned for the end of this year. The objectives of these tests include the following aspects with respect to slalom and side step manoeuvers:

- influence of rate command and attitude command parameter variations and
- influence of interaxis couplings.

Both, on-axis rate command and attitude command and the off-axis coupling response will be modelled in the frequency domain in order to investigate the required gain and the dynamic behavior.

5. CONCLUSIONS AND RECOMMENDATIONS

The application of rotorcraft in-flight simulators to support new helicopter designs has increased significantly in the past several years. This has been associated with the establishment of new flying qualities criteria reflecting the expanded role of helicopters and the application of new technologies.

In addition, the support of these general-purpose research vehicles during helicopter system and subsystem design process has been growing.

In this paper some recent DFVLR activities and results in the field of rotorcraft in-flight simulation were presented allowing to draw the following general conclusions:

- General-purpose in-flight simulators will play a decisive role in the rotorcraft design, evaluation, development, and certification process.
- The capability of in-flight simulators will support industry to meet the demands for future rotorcraft systems and to reduce costs and risks of development programs.
- For the generation of generic flying qualities data test vehicles providing adequate mission performance and variable characteristics are required.

Near future activities at DFVLR include the improvement of system capabilities of ATTHeS by implementation of additional new sensors, programmable cockpit displays, and variable controller characteristics.

For mid-term programs a new general-purpose in-flight simulator HESTOR is envisaged by DFVLR and MBB. The new vehicle will cover future needs of rotorcraft design validation for industry and serve as an evaluation tool for advanced rotorcraft system and subsystem acceptance testing for users and authorities.

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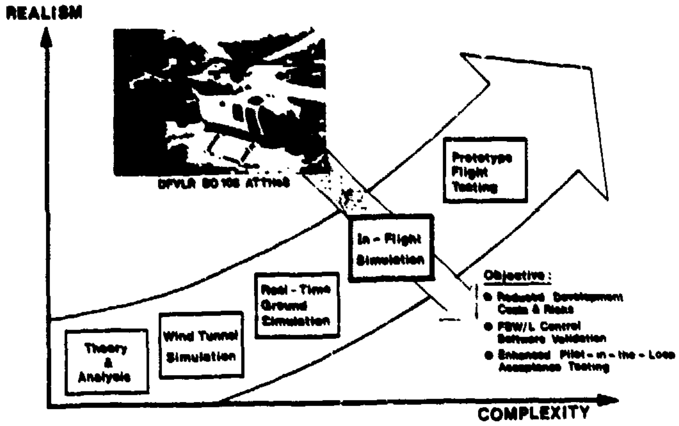

Figure 1. Role of In-Flight Simulation for Rotorcraft Design Validation

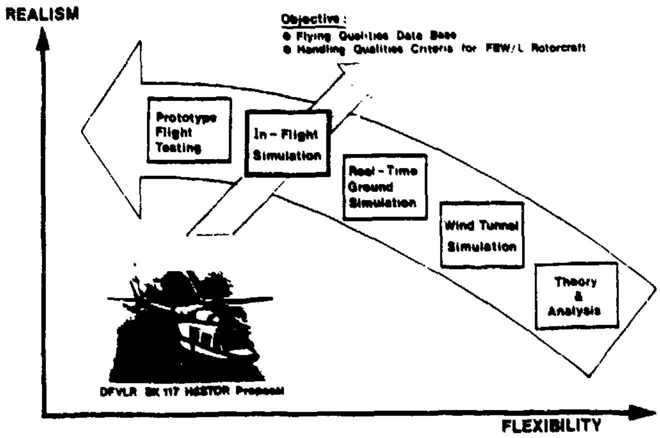

Figure 2. Role of In-Flight Simulation for Handling Quality Criteria Generation and Evaluation

Figure 3. In-Flight Simulator BO 105 ATTHeS

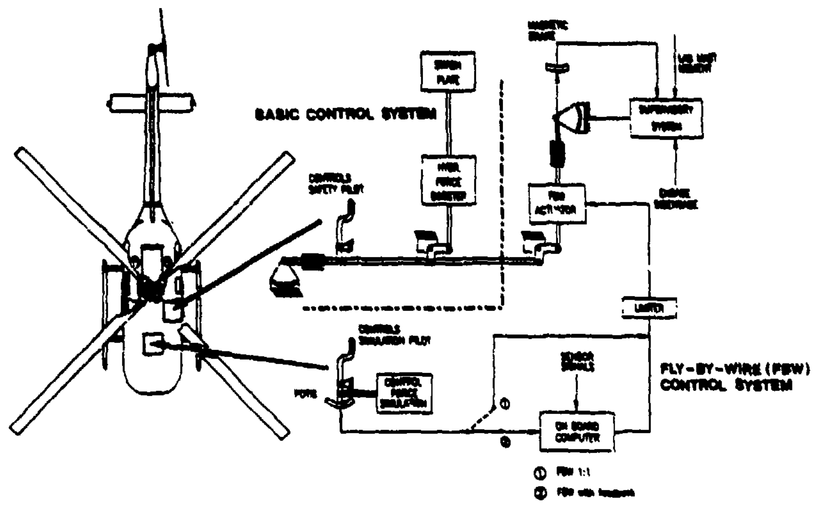

Figure 4. BO 105-S3 Control System Modification

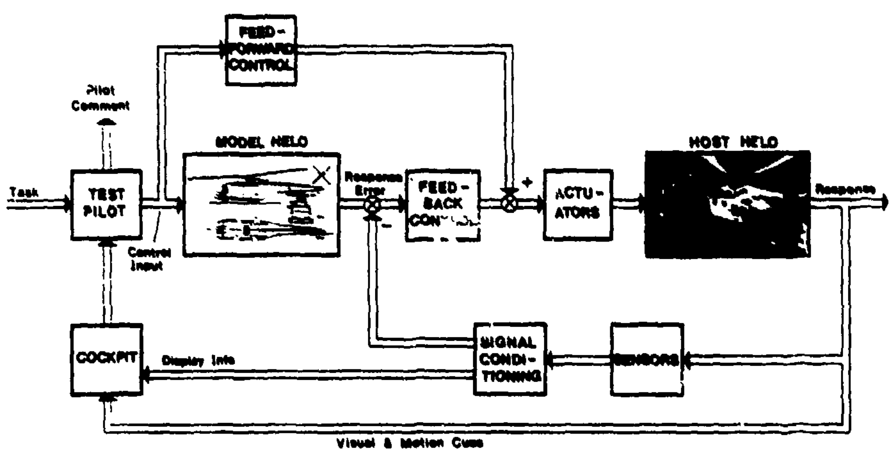

Figure 5. ATTHeS Model Following Control System

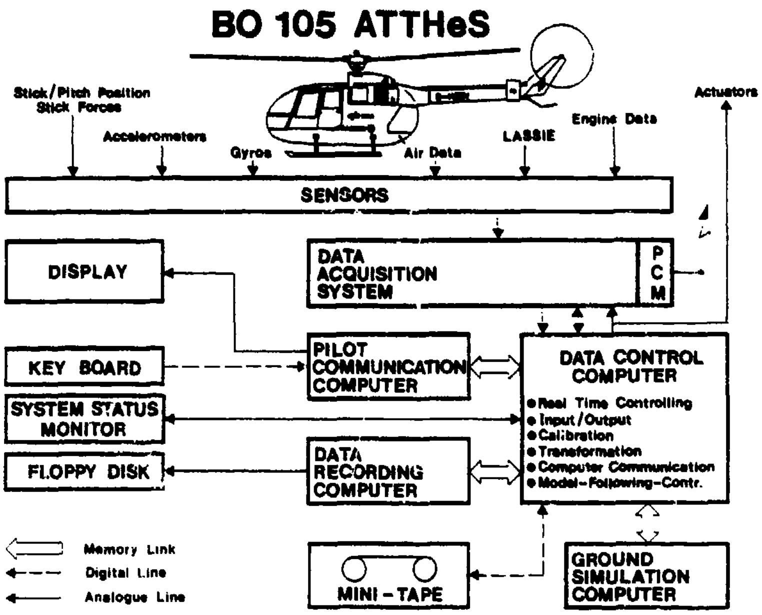

Figure 6. Realization of ATTHeS In-Flight Simulator

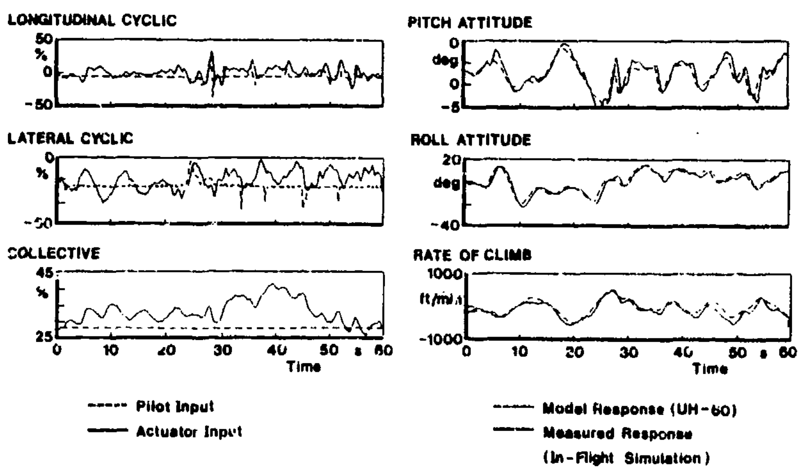

Figure 7. Model Following Control System Performance

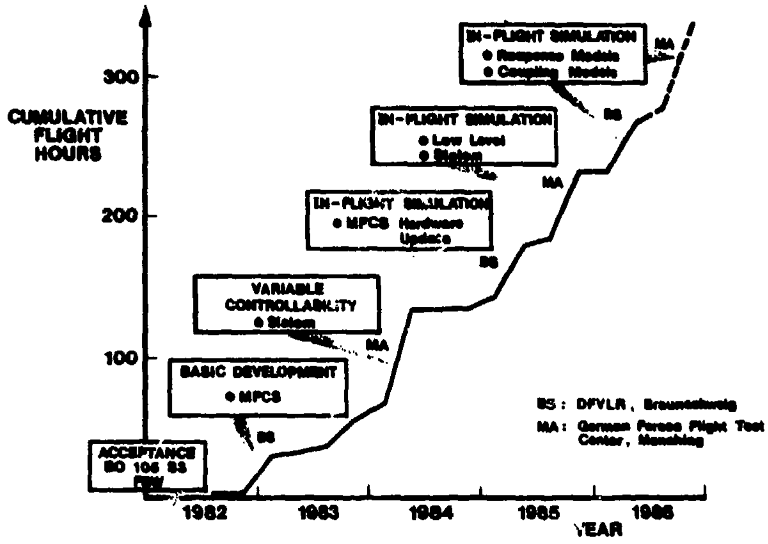

Figure 8. ATTHeS Flight Test Statistics

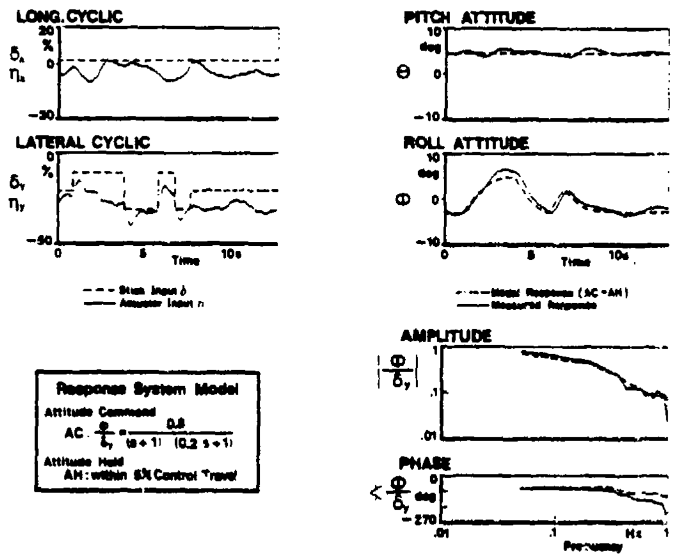

Figure 9. Realization of AC-AH Response System

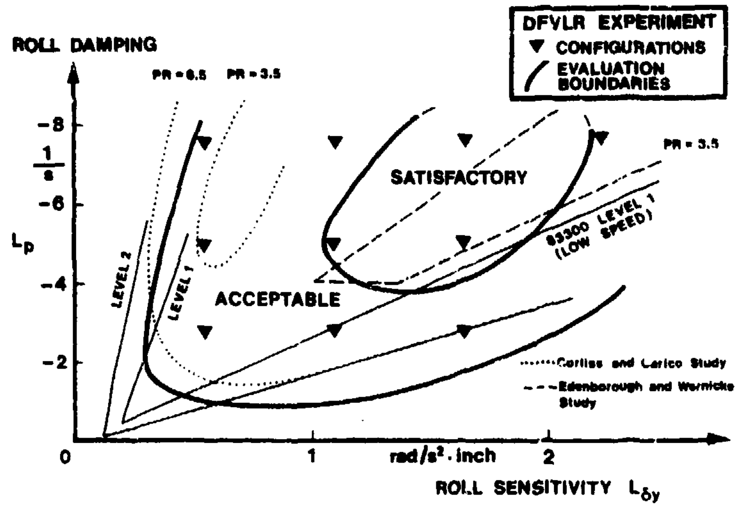

Figure 10. Roll Control Study Evaluation Diagram

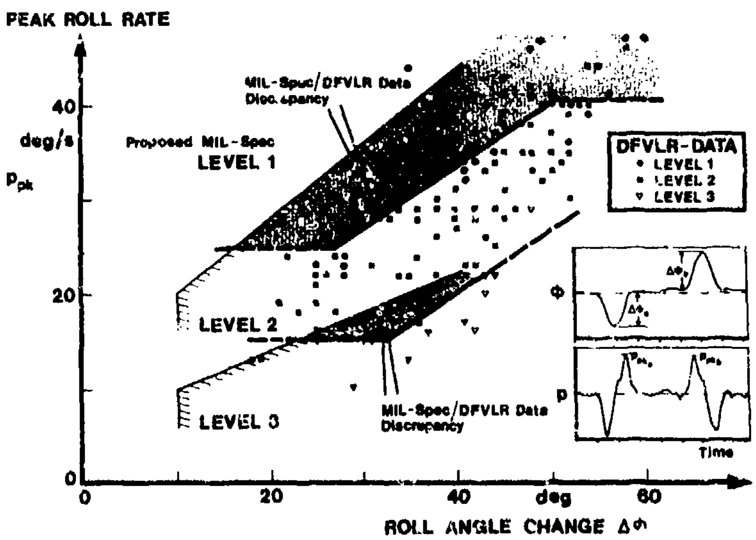

Figure 11. Roll Manoeuver Performance
- NOE Slalom Manoeuvers -

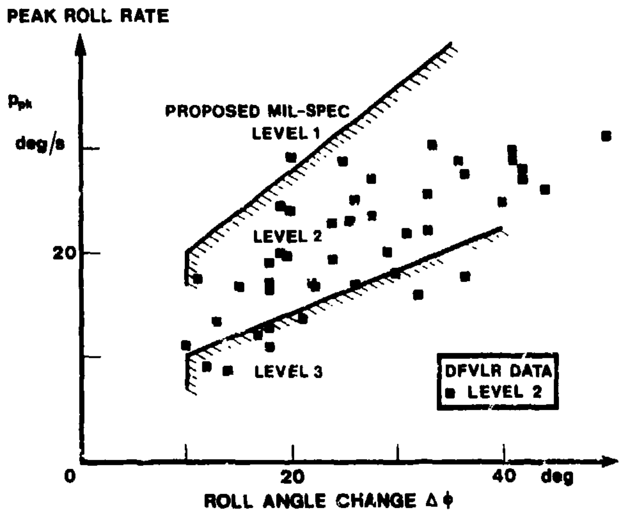

Figure 12. Roll Manoeuver Performance
- Side Step Manoeuvers -

INVESTIGATION OF VERTICAL AXIS HANDLING QUALITIES FOR HELICOPTER HOVER AND NOE FLIGHT*

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Summary

The preliminary results of two in-flight simulation programs on vertical axis rotorcraft handling qualities are presented. The parameters investigated in these studies were heave damping, Z_W thrust to weight ratio, T/W, and a number of dynamic engine governor/rotor system models. Flight tasks included hover, hover manoeuvring and nap of the earth (NOE) flight. Evaluation of 9 heave damping, thrust to weight ratio configurations provides the basis to suggest Level 1 Z_W and T/W boundary values of -0.20 sec^{-1} and 1.08 respectively. These boundaries are compared with other relevant work on the topic. The engine governor/rotor system evaluation results tend to disagree with handling qualities predictions based on a vertical velocity shaping parameter however further investigation is required.

Introduction

Vertical axis handling qualities have been the subject of numerous research programs. As early as 1962, in response to a VTOL handling qualities specification program, fixed-base simulations of VTOL tasks were carried out to determine handling qualities requirements for height control (Gerdes and Weick, 1962). The results of this initial study suggested an optimum height control sensitivity range of .21 - .37 g in^{-1} and a normal flight thrust to weight ratio (T/W) limit of 1.20. This work also pointed out the importance of aircraft heave damping (Z_w) and ground effect.

Further in-flight simulation by Kelly et al (1967) utilizing the NASA CH-47 showed that the T/W limit is strongly dependent upon the evaluation task. For a takeoff-circuit-landing task the minimum T/W for satisfactory flying qualities was placed at 1.09 within a minimum climb capability of 600 ft/min (3 m/sec). For an approach task alone, however, a minimum T/W of 1.03 could be allowed, provided that the heave damping, Z_w, was greater than -0.25 sec^{-1}.

Both of these studies indicated the strong coupling effects of collective (or lift control) sensitivity, $Z_{\delta c}$, heave damping, Z_w, and overall control authority, T/W. They also showed, not surprisingly, that task selection, and the inherent performance requirements for that task, can significantly affect any resulting parameter limits.

In 1979 Hoh and Ashkenas summarized the available data base for VTOL handling qualities and the data base deficiencies at that time. Some of the issues raised in that document include requirements to:

1) Establish the effects of external disturbances in longitudinal and lateral axes on vertical axis damping requirements, possibly by increasing longitudinal and lateral stability levels.

2) Isolate the effects of vertical axis damping from steady state climb rate due to collective input, $(w/\delta c)ss$.

and,

3) Investigate the effect of nonlinear engine response on handling qualities.

Renewed emphasis on vertical axis handling qualities, partially in response to the U.S. Military Specification 8501-Helicopter Flying Qualities (Reference 4 & 5) update program provides a strong impetus to further investigate the issues raised in these earlier research programs.

The Flight Research Laboratory of the National Aeronautical Establishment, under the auspices of TTCP and in close cooperation with Systems Technology Inc. and the U.S. Army, have been performing helicopter handling qualities research to support the ongoing 8501 update. The overall thrust of this program has been to provide in-flight simulation validated 'anchor points' corresponding to models used in fixed- and moving-base simulator research. This paper will deal with the segment of the Flight Research - 8501 program which is concerned with vertical axis handling qualities and will attempt to resolve the issues raised by earlier studies of the topic. This paper will describe two major experiments, one dealing with required heave damping levels in the environment of advanced control systems and the second dealing with the issues of T/W limits and the effects of non-linear engine governor/rotor system response. The paper will summarize the procedures and results of the first experiment (Reference 6) and will then discuss the preliminary results of the second experiment which is just drawing to a close. Both of the experiments have counterpart programs on the NASA-Ames VMS (Vertical Motion Simulator) and some research in this area is also being done on the NASA-AMES CH 47. The discussion of results here will be made in the context of these other two ongoing programs.

*This research was partially funded by the Canadian Department of National Defense.

Experimental Design

Both experiments to be discussed here were conducted on the NAE Bell 205 Airborne Simulator (References 7 and Figure 2). The evaluation pilot station was fitted with conventional controls with the characteristics shown in Table 1. As with all other in-flight simulations at NAE, the safety pilot was responsible for all system monitoring and overall flight safety.

The control systems used in this experiment wre all evaluated in an earlier Flight Research 8501 program (Reference 8). The three systems used were Attitude Command/Attitude Hold (ACAH), Rate Command/Attitude Hold (RCAH), and Rate Damped (RD) in pitch and roll axes. Each was designed level 1 Cooper Harper ratings for hover/manoeuvring tasks and control sensitivities were optimized by evaluation pilots at the onset of this program. A block diagram of three systems is included as Figure 2 and each control system bandwidth is listed in Table 2.

The yaw axis control system was a Rate Command system for all three pitch and roll systems and incorporated turn coordination and heading hold.

The vertical channel was implemented using one of two different approaches. For the first experiment, which utilized ACAH or RCAH in pitch and roll, the vertical channel modelling was an augmentation of the original Bell 205 dynamics. This implementation, shown in Figure 3, incorporates an aircraft vertical velocity (w) feedback loop to alter the heave damping of the aircraft. Modelled to first order as:

$$\frac{w(s)}{\delta c(s)} = \frac{Z_{\delta c}}{s - Z_w - K_f Z_{\delta c}}$$

the effective heave damping can defined as:

$$Z_{w\ eff} = Z_w + K_f \cdot Z_{\delta c}$$

For this implementation, K_f was varied to obtain $Z_{w\ eff}$ values of -0.05, -0.65 and -1.25 per second in hovering flight; -0.50, -0.80 and -1.10 per second in forward flight at 40 knots (20 m sec^{-1}); and -0.60, -1.10 and -1.50 per second in forward flight at 80 knots (40 m sec^{-1}). For hover and hover manoeuvring tasks $Z_{\delta c}$ was held at 0.28 g in^{-1}. Forward flight $Z_{\delta c}$ was initially also held at 0.28 g in^{-1} and later was used as a pilot controllable variable. The above quoted characteristics were calculated by hand from step collective input data and by a maximum likelihood parameter estimation package (Reference 9).

The second vertical channel model implementation was used in conjunction with the rate damped pitch and roll control system in the scond experiment. As depicted in Figure 4, this implementation is a model-following type in which Z_w, T/W, engine governor dynamics, ξ and α, rotor inertia, I_r, and collective/engine governor feedforward gain, G_2, could be easily varied. Use of this model was always coupled with a pilot adjustable collective sensitivity. The full transfer function of this model $w(s)/\delta c(s)$, is nonlinear in rotor rpm. Making small perturbation assumptions, and assuming perfect model following, results in the simplified transfer function of:

$$\frac{\Delta w(s)}{\Delta \delta c(s)} = \frac{g_1(G_1 K_1 \Omega^2 + 2K_1 \Omega \delta_c G_2) + I_r K_1 \Omega^2 s - 4K_1 K_2 \Omega^3 \delta_c}{M(SI_r - g_1 G_1 + 2K_2 \Omega \delta_c)(s + Zw)}$$

where: $g_1 = \dfrac{\alpha^2}{s^2 + 2\xi\alpha s + \alpha^2}$

Ω, δ are steady state values

M = aircraft mass

I_r = rotor inertia

K_1, K_2, h_1 are system gains

The values of Z_w, T/W, ξ, α, Ir and G_2 used in this experiment are listed in Table 3.

Experiment 1 - Heave Damping Effects

Utilizing RCAH or ACAH control systems in pitch and roll and the heave damping augmentation scheme in the vertical channel, evaluation pilots flew the hover/manoeuvring course depicted in Figure 5 and a preliminary nap of the earth (NOE) course. All configurations, three Z_w levels (-.05, -.65 and -1.25) for each of two control systems (ACAH and RCAH) were flown a minimum of three times prior to rating via the Cooper-Harper scale (Reference 10). Pilot comments regarding system deficiencies, pilot compensation requirements, performance and overall controllability were solicited and used to guide the overall analysis.

Hover course results for RCAH and ACAH control systems, shown in Figures 6 and 7, demonstrate the same major characteristic. The aircraft model with the lowest heave damping, -.05, was rated as possessing Level 2 flying qualities with a mean rating of 4.0 for all evaluations. The higher heave damping value models were all considered Level 1 aircraft.

The NOE results, based on flying the aircraft over tees and clearings in a forest while minimizing visual exposure (known as dolphin flying), were marred by variable weather conditions and a poor sampling of configurations. While the Cooper-Harper ratings were far too variable for analysis, pilot comments did provide good insight into the overall situation. Flights with fixed collective sensitivity (@ .28 g in^{-1}) resulted in the low heave damping cases being preferred over higher damping models. This phenomenon was attributed to the reduction in steady-state climb rate for a given collective input when heave damping is increased but collective sensitivity is kept constant, as shown in Figure 8. Later flights, where the pilot was allowed to vary collective sensitivity, showed that pilots preferred higher heave damping values when

steady-state performance was not compromised. This effect corresponds to the collective step responses shown in Figure 9, where the ratio of $Z_{\delta o}/Z_w$ is kept constant. In each flight evalution, pilot selected $Z_{\delta o}$ tended toward the constant $Z_{\delta o}/Z_w$ ratio. Pilot ratings during a later phase of hover course evaluation when collective sensitivity was 'pilot selectable' did not noticeably differ from earlier hover results.

The first experiment led to the following conclusions:

1) The effect of Z_w on helicopter handling qualities is less dramatic than the currently proposed 8501 update suggests. (It proposes a Level 1 limit of $-.25\ sec^{-1}$ and a Level 2 limit of $-.17\ sec^{-1}$ for all mission task elements other than dolphin and slalom tasks.)

2) The Z_w variation induced change in steady state rate of climb per inch of collective input is a strong effect when the task involves significant changes in rate of climb, such as the NOE task considered here. Tasks involving more stabilization in the vertical axis with few large changes in steady-state climb rate are unaffected by this same change.

Further examination of this initial experiment raised the following questions:

- The evaluation were flown with a relatively unrestricted T/W. Based on the temperature and aircraft weight a T/W on the order of 1.3 was not unlikely. Would the results of this experiment be valid at lower T/W's? What was the overall impact of T/W?

- Could the discrepancy between the results and the proposed 8501 heave damping limits be due to the stabilization level in pitch and roll axes provided by the RCAH or ACAH control systems, or were the 8501 limits just too conservative?

Experiment 2 - T/W, Zw and Engine Governor/Rotor Dynamics

To provide the best possible evaluation environment, the main task for this experiment differed from Experiment 1. Located in a marshy, uninhabited area, the new nap of the earth course (Figure 10 was able to incorporate all task elements of both courses used in the previous experiment except for the landing task. This NOE course started from the hover. The evaluation pilot then accelerated the helicopter to 40 knots (20 m/sec) and, while maintaining airspeed to within 5 knots (2.5 m/sec), 'dolphin' flew the aircraft over the three tree lines on the course. A rapid deceleration into hover followed and, after a short hover period, the pilot flew a bob-up to acquire and sight on a target. The pilot was required to have the target stabilized in the aircraft sight for 2 seconds and to be in visual line-of-sight contact with the target for no more than 4 seconds. After the descent from the bob-up the evaluation pilot was required to 'pedal turn' 180 degrees, rapidly accelerate to 40 knots and return to the start point following the same dolphin course outbound. Landings were evaluated separately at the NRC infield. Each configuration was flown through the course a minimum of three times or was landed a minimum of five times before an evaluation form was filled out. Variables evaluated in this experiment were heave damping (Z_w), thrust to weight ratio (T/W), and engine governor/rotor dynamics. A matrix of 3 Z_w values ($-.65$, $-.30$, $-.05\ sec^{-1}$) and 3 T/W values (1.1, 1.05, 1.03) formed a majority of the configurations flown.

The thrust to weight variable was incorporated into the experiment by feeding the simulated engine torque, Q_E, (Figure 4) to a cockpit gauge. This signal was conditioned to place the torque required for the model maximum T/W at the gauge redline for each T/W selected. Note that this T/W implementation is a steady-state criteria, not an unsteady incremental acceleration criteria as specified in the proposed 8501 update. Aside from an easier implementation, this steady-state criteria was felt to be the more rlevant parameter.

Along with the torque gauge, evaluation pilots were given an aural torque cue. At 5% below redline an 8 hz beeper commenced, at redline the beeper changed to a solid tone. Flight requiring torque at redline (solid tone audio), either momentarily or continuously, was considered "adequate performance not attainable" - a Level 3 Cooper Harper Rating.

Five engine governor/rotor models were selected for evaluation (Table 3). These models ranged from a very benign system quite similar to the original Bell 205 system (No. 0) to a torque overshoot prone model (No. 2), to a very slow governor response system (No. 6). Some sample time histories of these models are included as Figure 11. For evaluations interested purely in T/W and Z_w effects, mdoel No. 0 was implemented. The other dynamic models were all evaluated at T/W = 1.10 and Z_w = -65. Som combinations of reduced T/W, reduced Z_w and engine governor/rotor dynamics were also evaluated.

Experiment 3 - Preliminary Results

Preliminary results of the T/W, Z matrix evaluation (with dynamics model No. 0) are shown in Figue 12. Initial screenings of the subject data show that:

a) The bob-up and dolphin tasks consistently appear to result in the most critical handling qualities ratings. While the quickstop ratings also appear critical, these ratings are currently suspect due to a recently found modelling error when the aircraft is at significant pitch angles.

b) For a constant Z_w of -0.65, a T/W value of 1.10 results in a Level 1 aircraft. On the other hand a T/W value of 1.03 in a Level 3 aircraft for the same Z_w value. Pilot comments went from "marginal peformance in the bob-up" to "insufficient performance for bob-up" for these extreme T/W values. Figure 13 shows typical bob-up climb performance for the three T/W models with $Z_w = -0.65$. In another case the evaluation pilot, when confronted with the poor bob-up performance of the 1.03 T/W model, tried to enhance the climb performance by 'off-loading' the tail rotor. This technique, common for low T/W helicopter operations at the hover, allows the normal tail rotor torque to be applied to the main rotor while the aircraft is allowed to change heading in response to the main rotor applied torque. While this technique did not allow

increased performance of the aircraft due to the model implementation used here, its use does demonstrate the severity of the performance problem for T/W = 1.03. For this specific example the aircraft model received a Cooper Harper rating of 8 for the bob-up task.

c) While the ease of vertical stabilization, and thus the level of heave damping is important, the ratings and comments for the task/configuration matrix of this study tends to show that available aircraft performance is an overriding factor. Therefore, a slight reduction in Z_w for low T/W helicopters which provides a greater climb capability, is preferred by evaluation pilots in some cases.

The models evaluated in this experiment had one significant difference from actual helicopters. No provision was made for simulating translational lift and therefore the models did not show performance gains with increasing forward speed. This situation, graphically represented in Figure 14 means that a real helicopter with 1.10 T/W in the hover/bob-up speed range, will have a much larger T/W available in most forward flight conditions, such as those normal used for NOE flight. With this in mind, it is clear that the bob-up ratings provide the most critical evaluation of the T/W requirement and therefore the Level boundaries sketched in Figure 12 are based on bob-up ratings.

Sketched in Figure 15 are the handling qualities Level boundaries for Z_w and T/W derived from this experiment, the proposed 8501 boundaries, which appear slightly more conservative, and the boundaries generated by VMS. NRC and VMS handling quality ratings for bob-up manoeuvres, also shown on the figure, demonstrate some agreement but differ markedly at 1.10 T/W when Z_w is below -.65. This difference, which accounts for the major difference in the Level 1 boundaries derived by the two facilities, may be accounted for by two possibloe explanations:

1) The reduced level of visual cues in VMS for the top of the bob-up. Such a reduction might increase pilot workload and thus rating level.

2) A stronger insistence on height stabilization in the VMS bob-up stressed pointing accuracy. This task difference could easily alter the signfiicance of low heave damping values.

Evaluations involving more complex engine governor/rotor dynamics have so far provided mixed results. Based on the majority of pilot opinions, ratings and behaviours, the following comments can be made:

1) In the context of these evaluation tasks, at these low levels of T/W, it appears that the short term torque behaviour dominates pilot opinion. Fast acting governors with torque overshoots are strongly disliked, as are lower rotor inertia systems. The typical rpm droop model with a slow governor (No. 6) however, was well liked, probably due to the higher vertical accelerations allowed by trading rpm for thrust and the slower torque response. In this series of experiments it must be noted that although rotor rpm was indicated on a gauge with specific operating range limits, rotor rpm was not a closely controlled or evaluated variable.

2) When the torque dynamics become objectionable to the pilot and impinge upon torque limits, all evaluation pilots tended to reduce collective sensitivity to obtain greater precision in collective input. In one specific case this behaviour altered a landing rating from a 6 to a 3 with collective sensitivities of 0.29 and 0.16 g in^{-1} respectively.

When the pilot ratings are viewed in the context of the criterion proposed for the revised military specification, however, the interpretation is unclear. This criterion, based on work in this area by Hindson (1986) and Corliss (1983) suggests limits on the values of tr_{25} and a vertical velocity shaping parameter, $(tr_{50} - tr_{25})/(tr_{75} - tr_{50})$ where tr_n is the rise time of vertical velocity from the onset of a step input to n percent of the steady-state value. The results of this present study, when plotted in these terms (Figure 16) show no significant correlation. As discussed in Reference 11, the generation of the vertical velocity shaping parameter is very sensitive to slight anomalies in flight test daa and even a pure first-order dynamics model, which ideally is Level 1, can fail to meet the Level 1 criterion. With this in mind the dynamics models which have been used in this experiment are undergoing further investigation. It is hoped that frequency plane analysis techniques will provide the information necessary to clearly resolve the quantification and impact issues of complex engine governor/rotor dynamcis.

Conclusions

In conclusion, based on the data analysed to date, the following statements can be made:

- Level 1 limit envelope for T/W and Z_w suggested by this series of experiments is close to, but slightly larger than, the currently proposed 8501 limit envelope. Reasons for the discrepancies between the NRC and VMS envelopes still need to be addressed.

- This NRC derived envelope is driven primarily by bob-up task ratings.
- The results of the second experiment tend to suggest that for low values of T/W the provision of heave damping at the expene of steady-state climb rate is not desirable.

- Since the results of both the first and second experiments tend to agree, despite the reduction in pitch and roll stability due to the change from ACAH and RCAH to RD control systems, the effect of off-axes stabilization on vertical axis handling quality requirements is not discernible in this study. A further reduction is off-axes stability through the use of a more primitive control system should be evaluated for resolution of this question.

- Poor engine governor/rotor dynamics can, in some cases, be offset by a reduction in collective sensitivity.

- More analysis on the engine governor/rotor system dynamics models is required to allow insight into the strong pilot preferences and opinions expressed during that phase of the experiment.

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	Experiment 1	Experiment 2
Pitch		
stick force gradient	.5 lb $inch^{-1}$	.6 lb $inch^{-1}$
breakout	.5 lb	.9 lb
Roll		
stick force gradient	.5 lb $inch^{-1}$	.6 lb $inch^{-1}$
breakout	.5 lb	.5 lb
Yaw		
stick force gradient	6.5 lb $inch^{-1}$	6.5 lb $inch^{-1}$
breakout	1 lb	1 lb

TABLE 1: STICK CHARACTERISTICS

	Bandwidth Pitch	Bandwidth Roll	Overall Hover Course CHR
ACAH	2.74 rs^{-1}	3.10 rs^{-1}	2 - 2 1/2
RCAH	2.00 rs^{-1}	2.80 rs^{-1}	3
RD	1.80 rs^{-1}	3.50 rs^{-1}	2

TABLE 2: CONTROL SYSTEMS BANDWIDTH

Model No.	ζ	α	G_2	I_r (slug ft²)
0	.7	.2	10	1500.
1	.7	.5	10	200.
2	.7	.5	40	200.
4	.7	.5	40	1500.
6	1.0	.25	5	200.

TABLE 3. ENGINE GOVERNOR/ROTOR MODELS

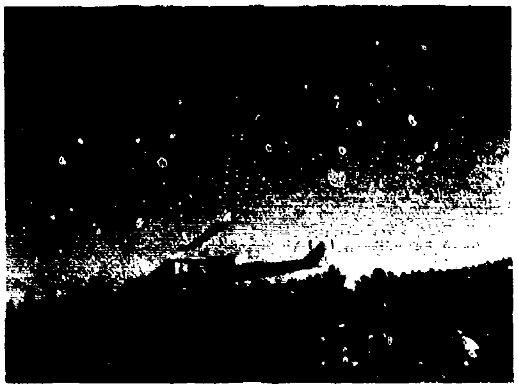

FIG. 1: THE NAE BELL 205 AIRBORNE SIMULATOR

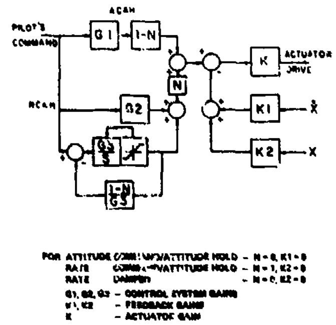

FIG. 2: PITCH AND ROLL CONTROL SYSTEMS

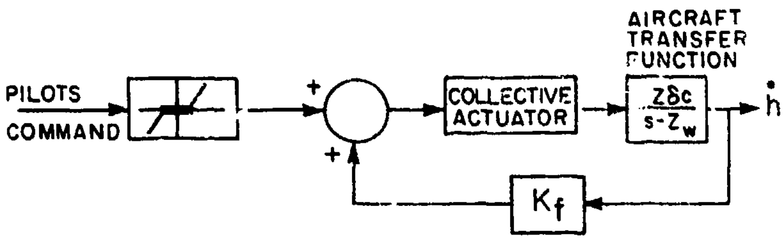

FIG. 3: HEAVE DAMPING AUGMENTATION SYSTEM - EXPERIMENT 1

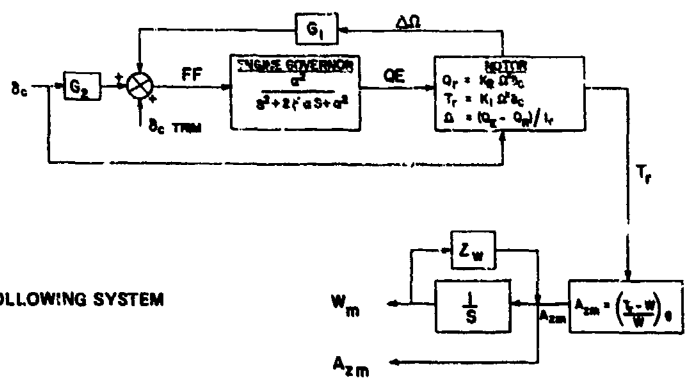

FIG. 4: VERTICAL CHANNEL MODEL - FOLLOWING SYSTEM EXPERIMENT 2

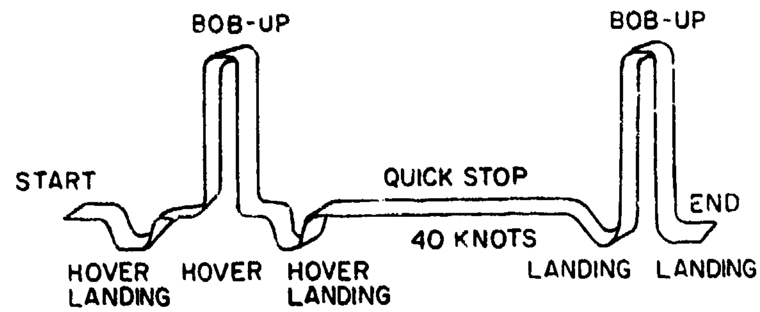

FIG. 5: HOVER COURSE - EXPERIMENT 1

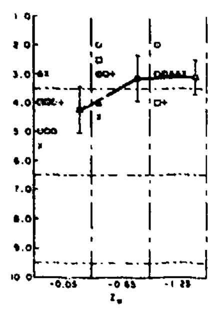

FIG. 6: OVERALL COOPER - HARPER RATINGS – HOVER COURSE, RCAH

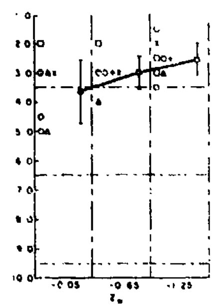

FIG. 7: OVERALL COOPER - HARPER RATINGS – HOVER COURSE, ACAH

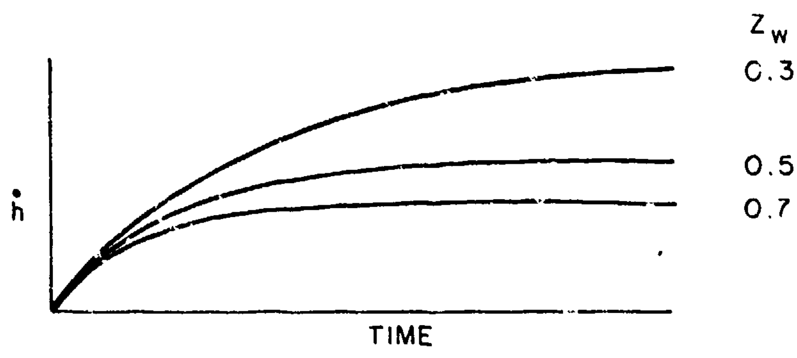

FIG. 8: STEP RESPONSE, $Z_{\delta c}$ FIXED

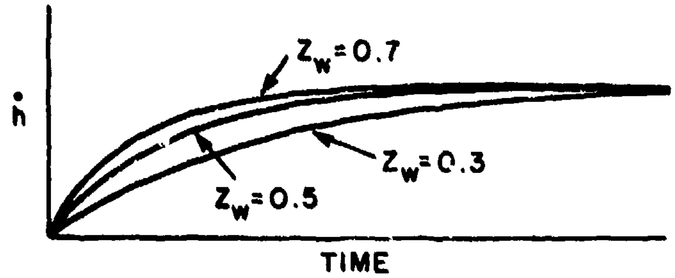

FIG. 9: STEP RESPONSE, $Z_W/Z_{\delta C}$ FIXED

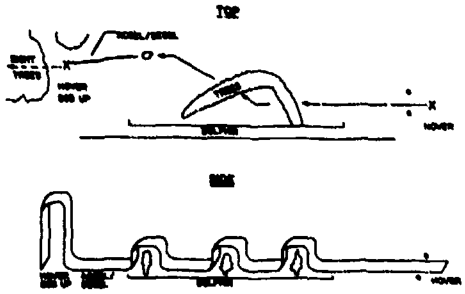

FIG. 10: NOE COURSE

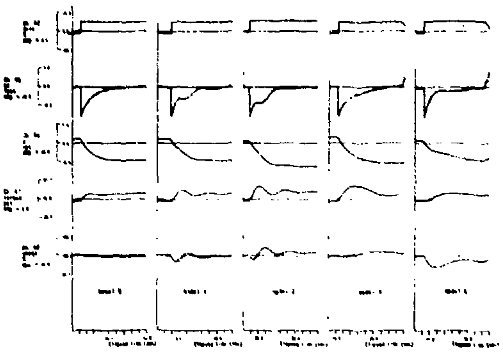

FIG. 11: ENGINE GOVERNOR/ROTOR MODELS

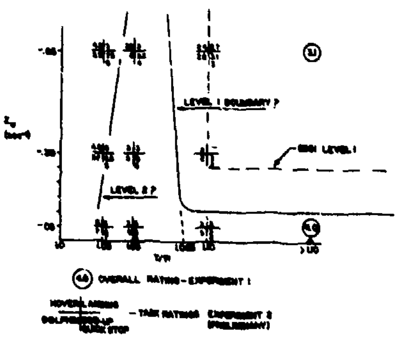

FIG. 12: THRUST TO WEIGHT/HEAVE DAMPING LIMITS

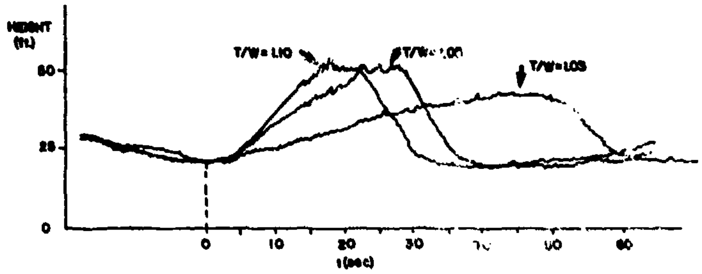

FIG. 13: EFFECT OF T/W ON BOB - UP PERFORMANCE

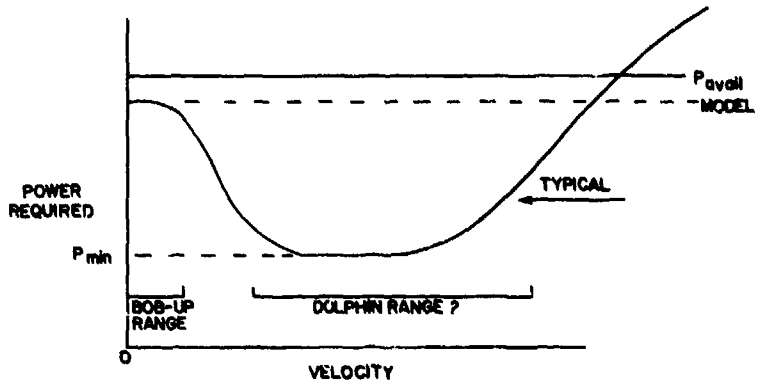

FIG. 14: TYPICAL POWER REQUIRED AND AVAILABLE CURVES

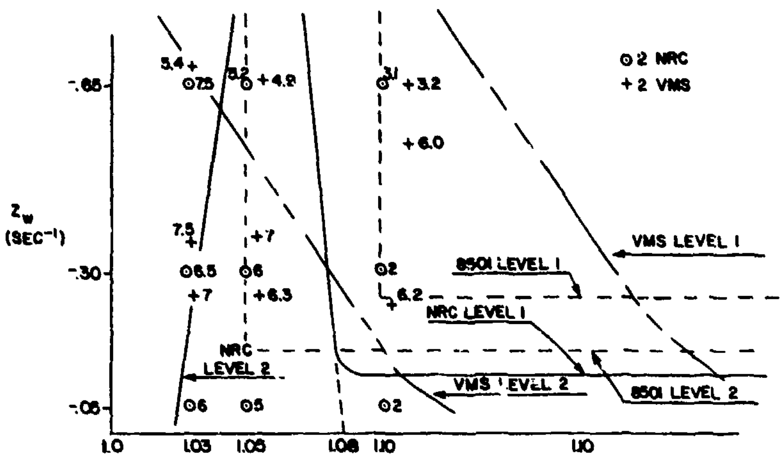

FIG. 15: T/W - Zw ENVELOPES — BOB - UP TASK ALONE

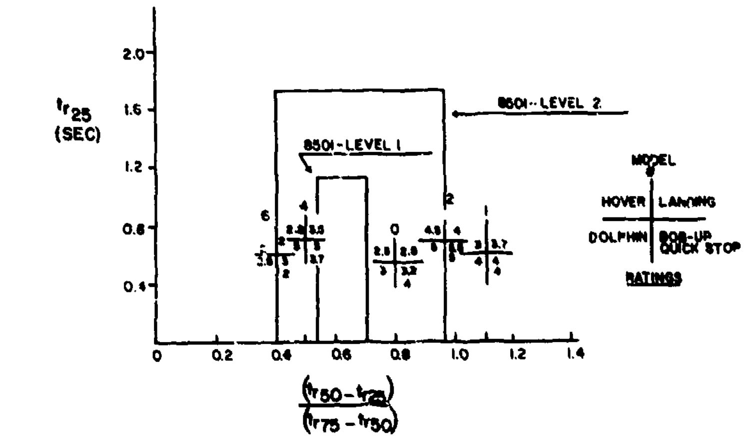

FIG. 16: PRELIMINARY ENGINE GOVERNOR/ROTOR DYNAMICS EFFECTS

HANDLING QUALITIES CRITERION FOR VERY LOW VISIBILITY ROTORCRAFT NOE OPERATIONS

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SUMMARY

The missions proposed for the next generation helicopter involve requirements to operate in essentially zero visibility in the nap-of-the-earth (NOE) environment. Such operations will require the use of pilot vision aids, which gives rise to the question of the interaction of such displays and the required aircraft handling qualities. This research was conducted to: 1) investigate the required visual cueing for low speed and hover, and 2) determine if an increase in stabilization can effectively be used to compensate for the loss of essential cues. Two flight test experiments were conducted using a conventional helicopter, and a variable stability helicopter, as well electronically fogged lenses and night vision goggles with daylight training filters. The primary conclusion regarding the essential cues for hover was that fine grained texture (microtexture) is more important than large discrete objects (macrotexture), or field-of-view. The use of attitude command augmentation was found to be effective as a way to makeup for display deficiencies. However, a corresponding loss of agility occurred with the tested attitude command/attitude hold system resulting in unfavorable pilot comments. Hence, the favorable control display tradeoff must be interpreted in the context that the best solution would be to improve the vision aid. Such an improvement would require an increase in the visible microtexture, an advancement in display technology which is unlikely to be available in the foreseeable future. Therefore, a criterion was developed to systematically evaluate display quality, and the associated upgrade in required stabilization as a function of increasingly degraded visual cues.

I. INTRODUCTION

The next generation helicopter must be able to operate at night, and in poor weather in the nap of the earth (NOE) environment to achieve adequate combat effectiveness. This gives rise to two critical issues, 1) collision avoidance with fixed objects, and 2) control and stabilization. In this paper, a criterion is developed specifically to address the control and stabilization issue for use in a revised rotorcraft handling qualities specification to supersede Mil-H-8501A (Ref. 1). The impact of displays on handling has never been accounted for in a handling qualities specification, and hence the proposed methodology is new and relatively untested. However, it is well supported by the theory of closed loop pilot vehicle analysis (Ref. 2), as well as data from two flight test experiments, and a ground-based piloted simulation (NASA Ames Vertical Motion Simulator). The criterion addresses the additional automatic flight control system (AFCS) stabilization that may be utilized to makeup for certain display deficiencies in the NOE environment. Improved displays which allow low workload NOE operations in very low (essentially zero-zero) visibilities, might someday obviate the need for such a criterion. However, such a quantum advance in display technology seems unlikely in the foreseeable future.

Both collision avoidance, and control and stabilization are addressed in the present version of the proposed specification revision which exists in the form of a U.S. Army Aeronautical Design Standard (ADS 33, see Ref. 3*). Collision avoidance is specified in Ref. 3 terms of three-dimensional maneuvering envelopes. The manufacturers are required to demonstrate that these envelopes do not fall outside the visual field of the available displays and/or vision aids. In the present paper however, we shall focus our attention on the development of a criterion for control and stabilization in the presence of degraded visual cueing.

II. BACKGROUND AND SUPPORTING THEORY

1. Development of the Specification Methodology

The proposed revision to the Ref. 1 specification will be heavily couched in automatic flight control system terminology in recognition of the fact that modern rotorcraft will utilize full authority fly-by-wire flight control systems. For example,

*The control and stabilization criterion presented herein has undergone some revision since Ref. 3 was published.

many of the criteria in ADS 33 (Ref. 3) are written in terms of "Response-Types" which classify the generic rotorcraft responses to control and disturbance inputs. The Response-Types defined in ADS 33 are Rate, Rate Command Attitude Hold (RCAH), Attitude Command Attitude Hold (ACAH), and Position Hold (PH). An incidental, but nontheless important byline to this is that the AFCS architecture is <u>not</u> specified. For example, the responses of a proposed acceleration command system were shown to fall in the ACAH Response-Type category.

In good visual conditions, most required tasks can be performed with a Rate Response-Type (see Ref. 3). In conditions of degraded visibility, and/or when the pilot must rely on vision aids, some of the cues required for control and stabilization are lost. The specification methodology is based on requiring additional AFCS stabilization (upgrade in Response-Type in spec terminology) in such conditions.

The basic elements required to carry out the proposed specification methodology are quantitative definitions of, 1) the Response-Types and, 2) the pilot's "usuable cue environment" (UCE). A viable definition for the UCE should include the following features:

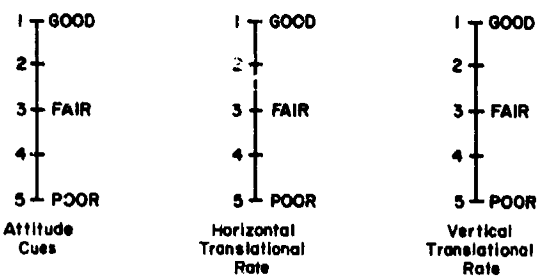
- It should depend on the pilot's ability to maneuver aggressively. In particular, it should not depend on the pilot's qualitative assessment of the usable cues. Experience gained during the Ref. 4 testing has shown that there is a strong tendency to overestimate the usefulness of available cues in a static environment.
- It should include the effects of all available vision aids and displays, including superimposed display symbology.
- It should not depend on the level of stabilization, since that is separately accounted for in the specification.
- Since quantitative metrics are not available, the UCE must be determined from a scale based on qualitative pilot evaluations. To the extent possible, the scale should:

  -- Utilize adjectival phrases with equivalent semantic meanings to all evaluation pilots.

  -- Be linear (e.g. a visual environment which is twice as bad should receive double the numerical rating).

  - Have low variability. Repeat evaluations, and evaluations for several pilots should result in rating scores with a low standard deviation.

The visual cue rating (VCR) scale in Fig. 1 was developed to satisfy these requirements. The words "good, fair, and poor" were shown to have low variability, to be linear, and to have essentially equivalent semantic meanings in the rating scale experiments described in Ref. 5. The definitions of cues given below the scales define maneuvering in terms of aggressive, moderate, and gentle corrections. These were developed from the pilot-vehicle analysis considerations presented in the following subsection, and have been tested and refined during the flight test experiments discussed herein.



<u>Definition of Cues</u>

X = Pitch or roll attitude and lateral, longitudinal, or vertical translation rate.

Good X Cues:	Can make aggressive X corrections or changes with confidence.
Fair X Cues:	Can make only moderate X corrections or changes with confidence.
Poor X Cues:	Only small and gentle corrections in X are possible, and consistent precision X control is not attainable.

Figure 1. Visual Cue Rating (VCR) Scale

2. Supporting Pilot-Vehicle Analysis Considerations

Performance of low speed NOE maneuvering requires that the pilot be able to perceive certain aircraft states with sufficient clarity to use them, and their derivatives, as feedbacks. For the conventional unaugmented helicopter, these feedbacks consist of aircraft attitude, and its derivative (angular rate), and aircraft position, and its derivative, translational velocity. This is illustrated in Fig. 2a, where the pilot is modeled according to conventional pilot-vehicle analysis (see Ref. 2). If attitude stabilization (attitude command attitude hold, ACAH) is provided, the block diagram in Fig. 2b would apply. The stabilization resulting from various combinations of pilot and/or SCAS equalization is summarized in terms of root loci in Fig. 3, which results in the following observations for a typical rotorcraft which may be characterised by the classical hover cubic.

- From Fig. 3a, it is not possible to maintain a stable hover without attitude stabilization.
- From Fig. 3b, closure of the attitude loop without lead, is conditionally stable, and is limited in terms of maximum achievable damping. The position loop closure requires considerable lead i.e., the translational rate cues must be good.
- From Fig. 3c, the use of lead in the attitude loop allows a much better inner loop around which to close the position loop. As a result, the position loop closure requires less lead i.e.,the translational rate cues only need to be fair. Note that the attitude loop lead carries into the outer position loop as a consequence of the assumption of a series pilot model (Fig. 2a).
- Figure 3d, represents the situation where ACAH augmentation is employed (Fig. 2b). A stable position loop closure is possible over a wide range of pilot gain, and the required position loop lead is only moderate i.e., only fair translational rate cues are required. This root locus also applies to the nonaugmented case, if a parallel pilot model structure is assumed (see Ref. 2).

The point to be made is that a good attitude loop closure alleviates the requirement for lead in the position loop. On this basis, an attitude command attitude hold SAS would be expected to compensate for degraded translational rate cues; a result which forms the foundation for the proposed criterion. The visual cue rating scale in Fig. 1 is intended to provide some measure of the available cues ("usable cue environment") for controlling attitude and position. The ability to develop attitude lead is believed to require high quality visual cueing. Based in the attitude root loci in Figs. 3b and 3c. the, lack of such lead results in a conditionally stable response, one which would preclude aggressive attitude corrections. Hence, the visual cue scale in Fig. 1 is based on the ability to make aggressive corrections in attitude. This is carried over to the definition of horizontal and vertical rate cues based on similar reasoning. It was found to be extremely important in the experiment described in Section III that, in making VCR evaluations, the pilots avoid qualitative assessments of a display or vision aid that deviates from the Fig. 2 definitions.

III. EXPERIMENTAL DATA

1. Visual Cueing Experiment

The fundamental visual cues required to perform low speed and hover maneuvering in the NOE environment are not well understood. Knowledge of these essential cues is required for the development of pilot displays for low or zero visibility operations. This flight test experiment (described in detail in Ref. 4) provides some insight into the necessary cues for control and stabilization and the results are summarized in this section as supporting data for the criterion to be developed subsequently in Section IV. The primary variables in the Ref. 4 flight tests were the field-of-view, the amount of visible macrotexture (large objects) and microtexture (fine-grained detail). Six different fields of view were tested, varying from a small (10 deg X 10 deg) forward looking window to large windows (see Fig. 4) which had essentially no restrictions to peripheral vision.

The visible texture was varied by conducting the tests over two marked courses (Fig. 5) on a dry lakebed, and by using special electronically fogged lenses to remove the visible microtexture (cracks in te lake bed). The scope of the experiment did not allow quantitative measurements of the fogged lenses in terms of the modulation transfer function (see Section V). An estimate of the pilots' visual environment with the lenses fogged was obtained from a standard eye chart (Landolt rings) set up at the test site. The pilot's vision with the lens fogged tested from 20/20 to 20/40, even though the pilots generally agreed that the cracks in the lakebed were removed as usable visual cues. The details of visual cueing are discussed in Section V, where it is shown how it is possible to test 20/20 on a standard eyechart and still not be able to utilize small detail as a usable cue due to inadequate depth of modulation.

The visual cue ratings (VCRs from Fig. 1), and Cooper- Harper handling qualities ratings are plotted against the variations in field-of-view in Fig. 6. The following observations can be made from this data.

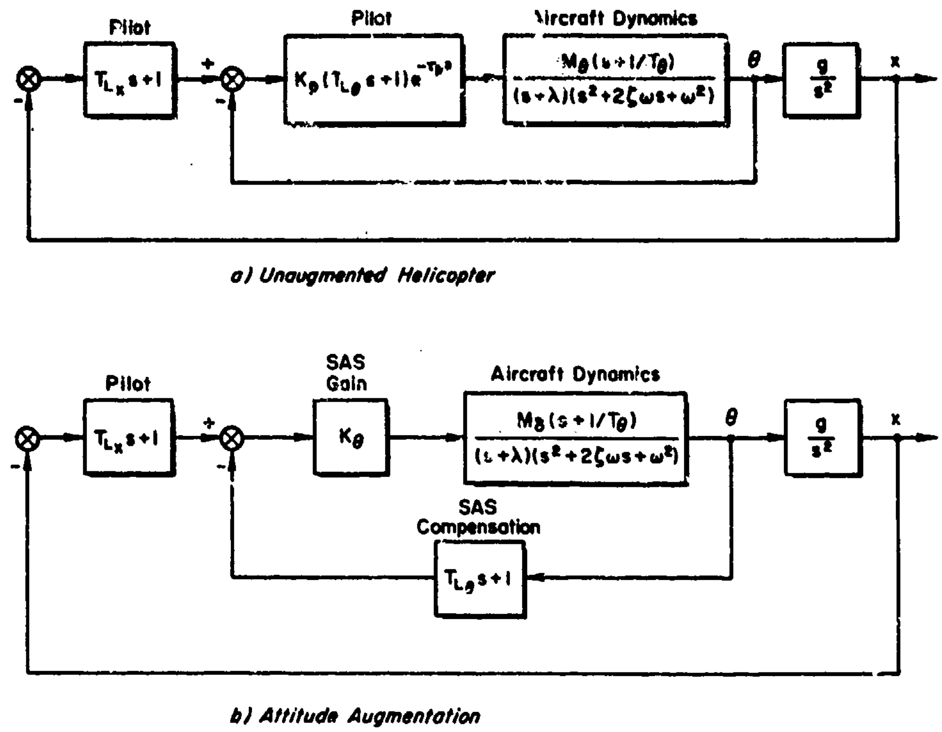


Figure 2. Piloted Closure for Hover

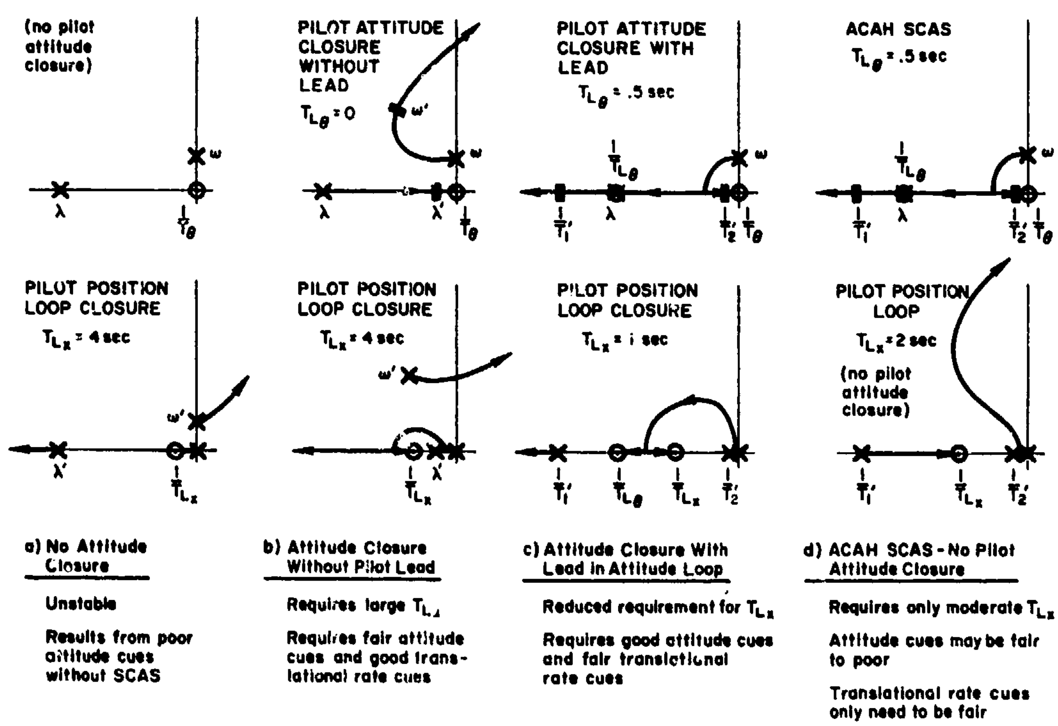
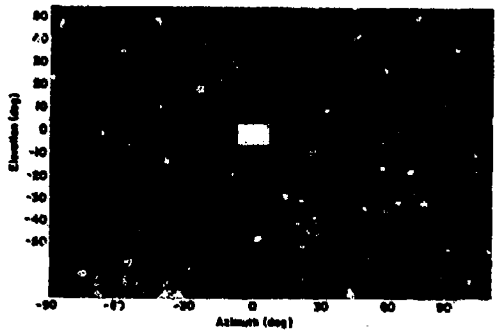
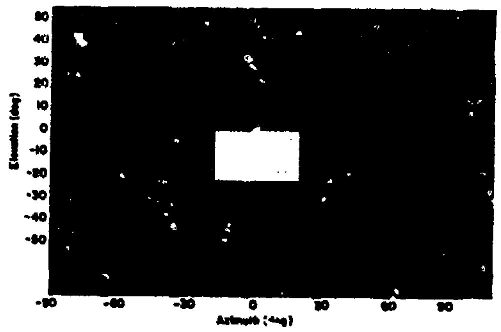


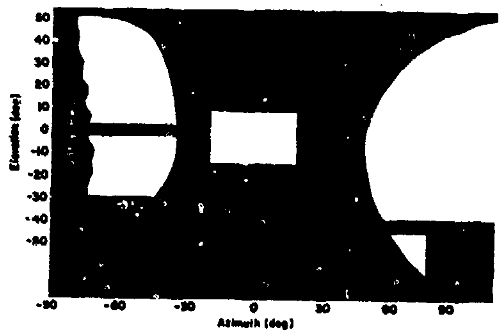
Figure 3. Tradeoff Between Attitude and Translational Rate Feedbacks Required for Stable Hover



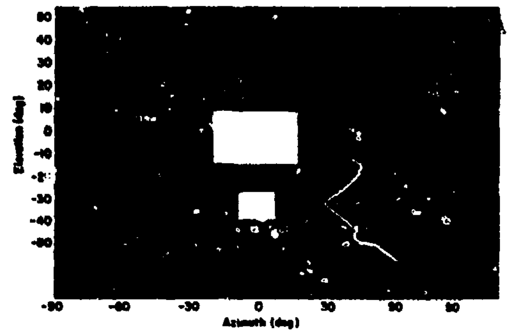
(a) Variations in field-of-view — configuration 1 (narrow upper front to evaluate potential displays)



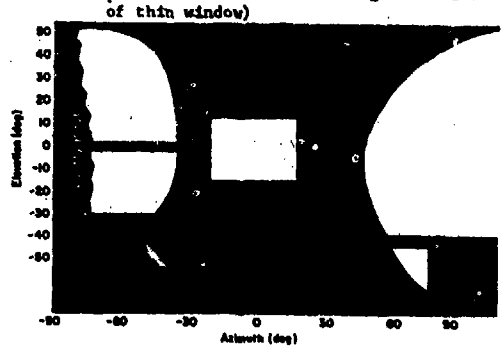
(b) Variations in field-of-view — configuration 2 (nominal upper front to simulate NASA Ames VMS front monitor)



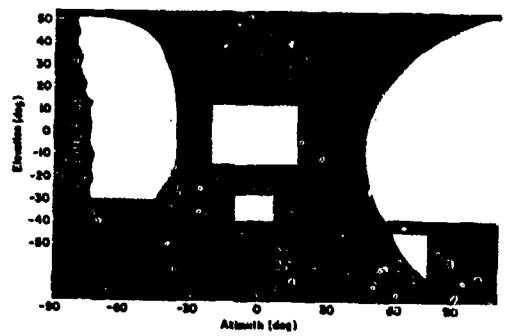
(c) Variations in field-of-view — configuration 6 (nominal upper front plus sides to investigate effect of thin window)



(d) Variations in field-of-view — configuration 3 (nominal upper front plus lower front to investigate effect of thin window)

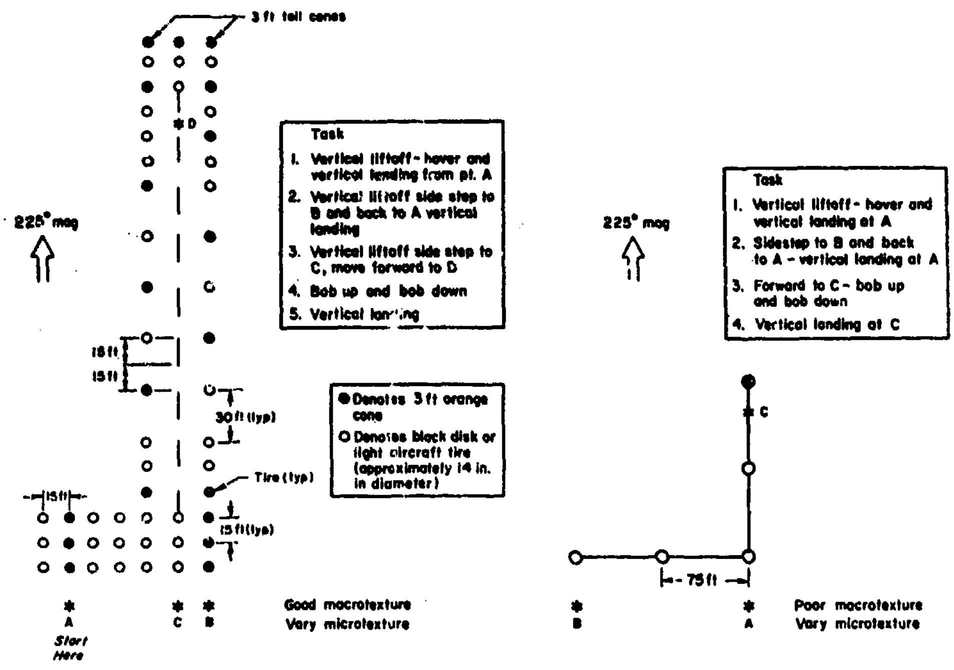


(e) Variations in field-of-view — configuration 7 (wide upper front plus sides)



(f) Variations in field-of-view — configuration 8 (wide upper front + lower front + sides)

Figure 4. Field-of-View Variations in Reference 4 Experiment



a) Test Site One

b) Test Site Two

Figure 5. Test Sites in Ref. 4 Experiment

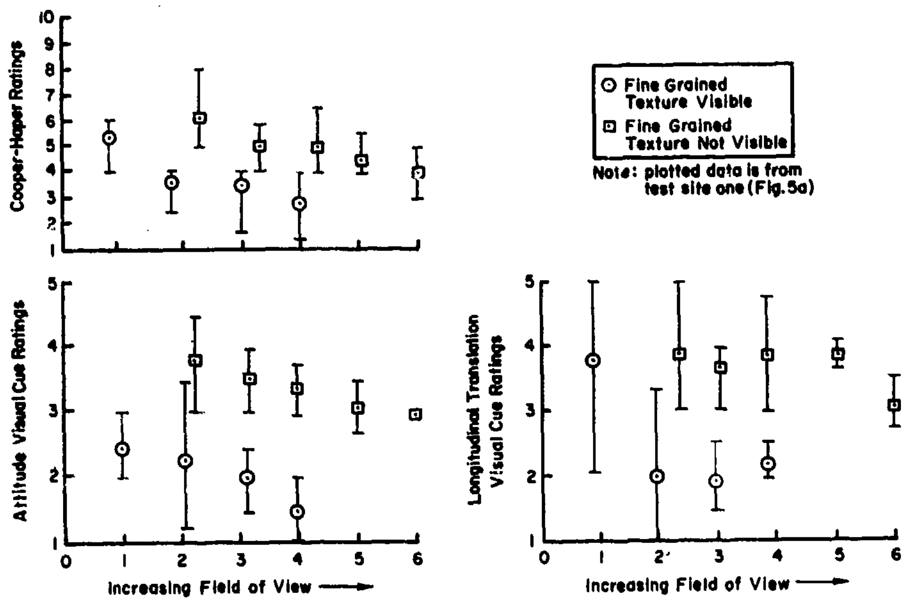


Figure 6. Effect of Field-of-View and Microtexture Variations in Ref. 4 Experiment

- Visible microtexture is an important visual cue for control and stabilization.
- Increasing the field-of-view beyond 38 deg X 23 deg (Configuration 2 in Fig. 4) does not result in significant improvements in Cooper-Harper or vsual cue ratings. An increase in field-of-view would, of course, be desirable for navigation and orientation, however, the results of this experiment indicate that it would be undesirable to increase the field-of-view of a pilot vision aid at the expense of resolution (visible microtexture).

The Fig. 6 data includes only the results obtained on Test Site one (Fig. 5a) which was rich in macrotexture. The results from Test Site two (Fig. 5b), which was devoid of macrotexture, were essentially the same. Hence, macrotexture was found to be of secondary importance to microtexture in terms of cues required for stabilization and control.

The averaged visual cue ratings are plotted against the averaged Cooper-Harper ratings in Fig. 7 to examine the effect of degraded visual cueing on handling qualities. These results show that:

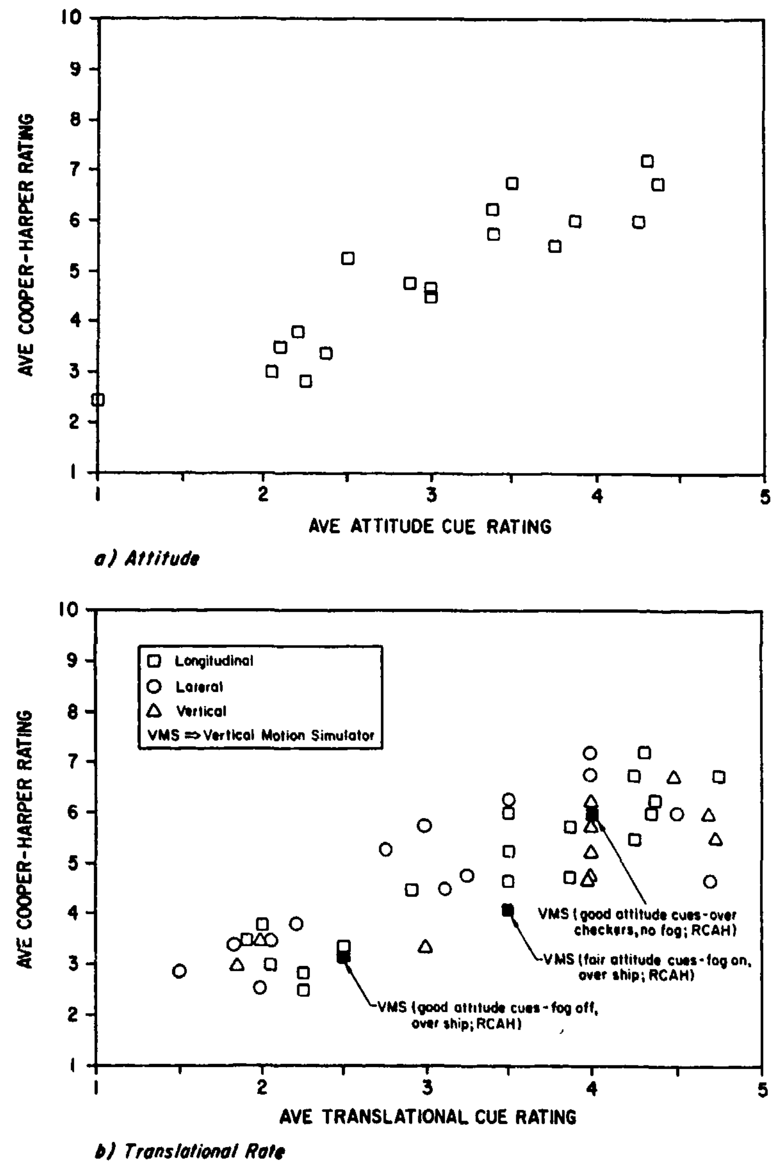


Figure 7. Variation in Average Visual Cue Rating For Attitude and Translational Rates With Average Cooper-Harper Rating (Data from Ref. 4 and 8)

- The test helicopter, a Hughes 500D, was given Level 1 ratings when the VCRs were 1.5 or better.

- The handling qualities ratings steadily degraded as the VCRs increased, which validates the trend predicted from the analysis in Fig. 3.

Some visual cue ratings were taken from a moving base simulation conducted on the NASA Ames VMS and these are also plotted in Fig. 7b. These data show that the trend of the degradation in Cooper-Harper ratings with increasing translational rate VCRs agrees reasonably well with flight test. The attitude VCRs were judged to be good for all cases on the simulator, indicating that the trends in Fig. 7b are not dependent on simultaneous degradation of attitude and translational rate cues.

The VCR scale (Fig. 1) not only plays a significant role in the proposed criterion, it also provides a quantitative metric for comparison of competing displays or vision aids. As noted earlier, the validity of such a scale depends on its ability to produce ratings with low variability within and amongst pilots. The variability of the Fig. 1 scale for the experimental data from Ref. 4 is shown in the cumulative distribution plot in Fig. 8, where the ordinate is the percentage of total VCR ratings with a standard deviation less than or equal to a given value on the abscissa. This data included over 200 separate evaluations. Based on this plot, it would be expected that the standard deviation in the VCR ratings in a given experiment would not exceed 0.75 more that 0.8% of the time. This is a reasonable validation of the scale, and is the basis for a Ref. 3 specification requirement that the standard deviation in the ratings not exceed 0.75. Such a deviation would be reason to suspect the existence of an anomalous set of ratings such as may be caused by a preconceived mind-set by one of the evaluation pilots. In such cases, the procuring activity may elect to assign additional pilots, or to make a decision based on other factors (such as eliminating one pilot's ratings, or emphasizing the pilot comments more than the numerical ratings).

2. Control-Display Tradeoff Experiment

An experiment was conducted to validate the analytically based hypothesis, that the addition of attitude stabilization would be effective as compensation for some loss in visual cues. If valid, such a hypothesis would allow for the possiblity that augmentation can be effectively utilized to make up for less than ideal displays and/or vision aids. This is especially useful in light of the fact that the fine-grained texture (microtexture), found to be an essential cue for stabilization and control in the above discussed visual cueing experiment, is very difficult to incorporate into displays and vision aids. For example, the usable microtexture is somewhat limited for forward looking infrared (FLIR) displays, computer generated imagery (CGI), and light intensifier systems or night vision goggles (NVG). This is further exacerbated by the tendency to increase the field-of-view at the expense of microtexture, a trend that improves positional awareness, but at the expense of control and stabilization.

The experiment discussed herein utilized a variable stability helicopter (Canadian National Aeronautical Establishment (NAE) Bell 205A), and night vision goggles as a representative pilot vision aid. The night vision goggles were a current state-of-the-art system (PVS-6). However, safety considerations dictated that they be used in the variable stability aircraft only in daylight conditions. Night conditions were simulated using variable density training filters which allowed the simulation of conditions varying from a full moon to a very dark night such as might exist in a rural area with a solid overcast. The following factors arise from the use of daylight filters to simulate the night environment.

- Pilots experienced with the PVS-6 night vision goggles indicated that they are much easier to use in the real night environment. Hence, the results of this study must not be used as a basis for an evaluation or comparison of the night vision goggles.

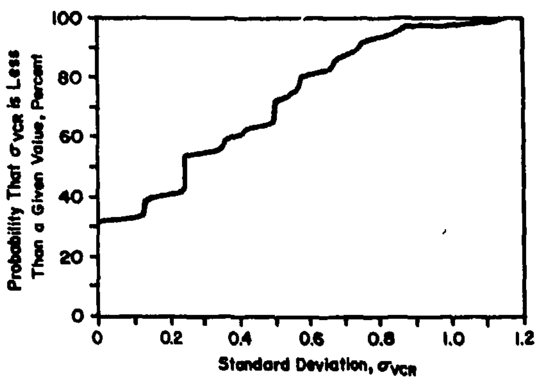


Figure 8. Cumulative Distribution of Standard Deviations of Visual Cue Ratings Given by Pilots

- The available texture depended greatly on the lighting conditions (overcast vs. sunny).
- Any direct glare from the sun severely degraded the visual scene.

Another factor which should be taken into account is the lack of available time to sufficiently train evaluation pilots to fly the night vision goggles. Most pilots were allowed about 2 hours of familiarization before conducting formal evaluations, which is substantially less than the time allotted by the U.S. Army to qualify for actual night vision goggle operations.

The variable stability Bell 205A was configured to simulate a rate augmented helicopter, and a helicopter with Attitude-Command-Attitude-Hold (ACAH) augmentation. Both configurations were tested to be Level 1 with no restriction to vision (Cooper-Harper handling quality ratings equal to or less than 3.5) in a previous handling qualities experiment (best Rate and ACAH systems from Ref. 6). The task was essentially identical to that used in Ref. 6 except that a bob up/down, and hover turn were added, see Fig. 9.

The test procedure involved setting the variable density filters, while sitting in the helicopter, at a calibration site wherein a standard eyechart was mounted 20 feet from the evaluation pilots head. The filters were run at two settings; wide open (pilots usually reported this as 20/70 in terms of visual acuity), and at a setting which resulted in a visual acuity of 20/85. To put this in an operational context, the PVS-6's tested between 20/50 and 20/60 on a full moon night, and about 20/85 on a dark overcast night wich some distant glow visible from airport runway lights (it was not possible to see the eyechart at all with the unaided eye).

Two test sites were utilized to further vary the visual environment. One site was over a large, flat, grassy field, and the other in a swampy area with large clumps of weeds which provided additional microtexture. Finally, tests were run with and without snow cover, on sunny and cloudy days, and on windy and calm days (most were calm).

A total of seven evaluation pilots participated in the experiment, although only four had enough familiarization time to achieve consistent, ratings. Visual cue ratings (VCRs from Fig. 1) and standard Cooper-Harper handling qualities ratings were obtained for the Rate augmented configuration, whereas only Cooper Harper ratings were obtained for the ACAH configuration. Visual cue ratings were not obtained for the ACAH cases because the use of such augmentation obviates the need for the critical cues. The pilots were required to fly the Fig. 9 test course at least three times before assigning the ratings, and recording their comments. This resulted in about 20 minutes of evaluation time which included 12 vertical landing, 3 sidesteps, quickstops, bobup/downs, turns about a point, and precision hovers. Separate Cooper-Harper ratings were given for each of these maneuvers.

The VCR and Cooper-Harper rating results were analyzed with a view toward answering the following questions.

- What is the interdependence between the three components of visual cues in Fig. 1 (attitude, horizontal and vertical translational rates)?
- To what extent does ACAH alleviate the degradation in handling qualities associated with a degraded visual environment?
- What combination of VCRs causes a Level 1 (Cooper-Harper 1 to 3.5) baseline Rate augmented aircraft, to become Level 2 (Cooper-Harper 4 to 6)?

PILOT TASKS

1. Precision Hover and Vertical Landing at A
2. Hover turn about A at constant radius
3. Rapid sidestep to B - stabilize while pointing at B' - and return to A
4. Repeat I
5. Quickstop to C
6. Bob up/down over C
7. Land at C
8. Return to A
9. Repeat 1-8 three times
10. Give ratings

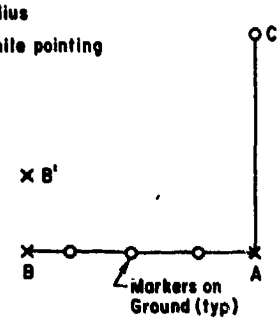


Figure 9. Test Course and Tasks Used in Night Vision Goggle Experiment (Also used in Ref. 6)

- Over that range of VCRs does the addition of ACAH upgrade the Cooper-Harper ratings from Level 2 (for the baseline Rate augmentation) to Level 1?

Each of these questions is addressed in the following paragraphs. Unless specifically noted, only data taken in calm conditions have been included in the analysis.

Interdependence Between Visual Cue Ratings

The interdependence between the three components of the Fig. 1 visual cue rating scale can be examined from the VCR rating data presented in Fig. 10 for Rate augmented cases with night vision goggles. Here it is seen that the vertical and horizontal translation cues are highly correlated (R^2 = .84) whereas the translation and attitude cues are relatively independent (R^2 = .38)*. A linear regression fit to the data is also plotted in Fig. 10. On this basis, the remainder of the analysis of the data is based on the attitude and horizontal translation cue rating (i.e., vertical translation cues are not included as an independent variable in the analysis).

Comparison Between Rate and ACAH In A Degraded Visual Cue Environment

The Cooper-Harper handling qualities ratings are plotted vs. the horizontal translation visual cue ratings for each of the tested maneuvers in Fig. 11. The following observations can be made from this rating data and the associated pilot commentary.

- The baseline Rate augmented configuration (triangles) exhibit a tendency toward increasingly inconsistent and degraded Cooper-Harper ratings with increasing VCR. This is consistent with the data from the visual cue experiment discussed in the previous section (see Fig. 7 and Ref. 4).

- Configurations with ACAH augmentation are given Cooper-Harper ratings between 3 and 4 up to a VCR of 4.5 for all maneuvers except the quickstop and bob-up/down. The quickstop received Level 2 ratings in the Ref. 6 experiment (no restriction to vision) due to the lack of agility inherent to the ACAH augmentation as mechanized. All pilots noticed problems in the bob-down with night vision goggles due to the lack of visible microtexture at altitudes above 10 to 20 ft. This resulted in a distinct lack of altitude and altitude rate awareness which was not alleviated by the ACAH augmentation (nor was it predicted to in the pilot-vehicle analysis in Fig. 3).

- The ratings for the ACAH cases, while better that rate cases in the degraded visual environment, did not reflect ideal conditions (i.e., Cooper-Harper ratings were 3 to 4). This might be improved with an optimized ACAH augmentation, however, it is suspected that the use of such augmentation is less attractive than restoring the visual conditions via improved displays. Even though ACAH allows the pilot to operate in degraded visual conditions, there is a distinct loss of aggressiveness due to the nature of ACAH, and to the above noted problems in the height axis. However, displays with adequate microtexture for stabilization and control, combined with an adequate field-of-view for positional awareness are not expected to be available in the near future. Hence, the use of augmentation to make up for display deficiencies represents the only compromise.

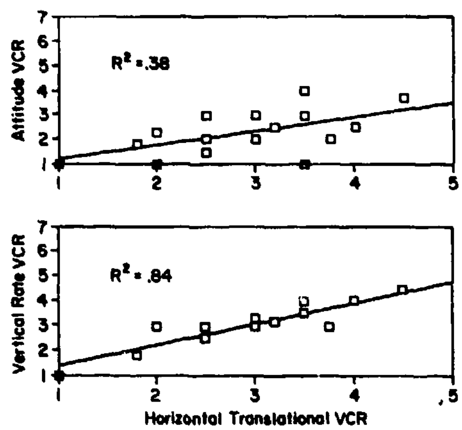


Figure 10. Interdependence between Visual Cue Ratings

*R is the correlation coefficient and $\sqrt{1 - R^2}$ = 0 represents perfect correlation.

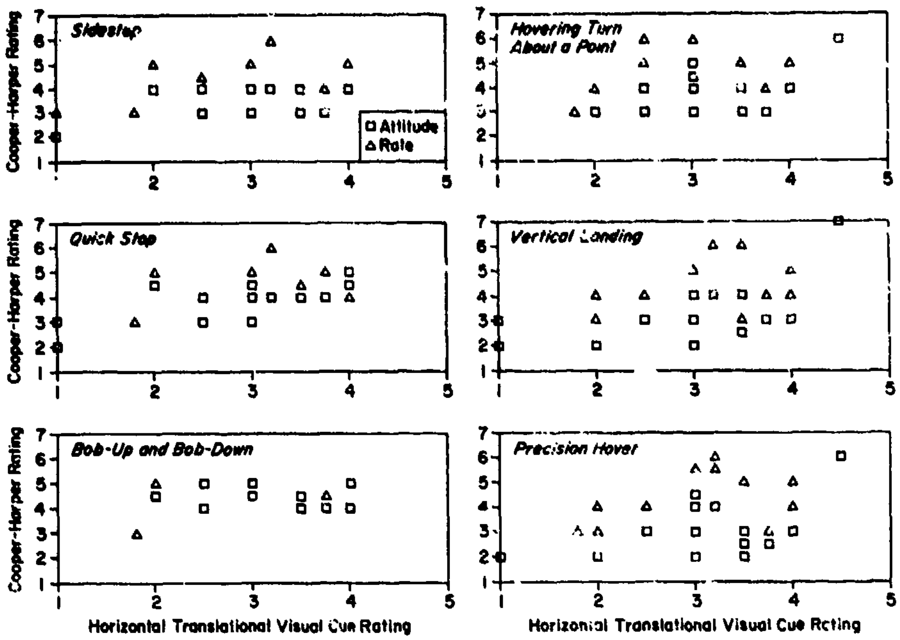


Figure 11. Cooper-Harper Ratings vs. Horizontal Translational Cue Ratings

The results shown in Fig. 11a through 11d effectively validate the basic hypothesis formulated in Fig. 3 (i.e. attitude augmentation can be used to offset degraded attitude and translation visual cues). The Cooper-Harper ratings for ACAH might have been even better if the stick force gradients were somewhat higher. This was noticed late in the tests as a result of continuing comments by the pilots that the ACAH case tended to "buck and shuffle" in response to pitch commands. It was then noted that the force gradient was a factor of four less than that used in a previous ground-based simulation (Ref. 8) conducted on the NASA Ames Vertical Motion Simulator (.5 lb/in in flight and 2.0 lb/in in the simulation). Increasing the gradients, and modifying the controller inertia and friction empirically, resulted in considerably improved pilot acceptance of the ACAH case. Interestingly, the lower stick force gradient was not noticed by 5 different pilots in the previous handling qualities tests (Ref. 6) suggesting that higher gradients are desired when degraded vision is a factor.

Effect of Visual Cue Ratings on Cooper-Harper Ratings

The effect of visual cue ratings on Cooper-Harper handling qualities ratings suggested by the experimental data was estimated by the application of a multiple linear regression.* This resulted in the following empirical relationship between handling qualities (HQR) and visual cues (VCR) for rate augmentation.

$$HQR = 0.89 + 0.89\ VCR_\theta + 0.60\ VCR_x$$

This regression fit was accomplished using the current experimental data for night vision goggles with Rate augmentation, and the data taken from the Ref. 4 experiment (discussed in III.2) resulting in a total of 89 observations. The correlation coefficient for this fit is .83 which statistically indicates correlation at substantially better than the 99% level of significance (Ref. 9). The estimated and actual ratings are plotted in Fig. 12, where it is seen that the data spread about the line of perfect correlation is reasonable up to ratings of about 7. Beyond this value, the linear fit is nonconservative. However, the complexity of a multiple nonlinear regression seems unwarranted, since only the data up to a rating of 6.5 is used in the subsequent criterion development.

*In the Ref. 4 experiment, the pilots gave a composite Cooper Harper rating for the full test course (Fig 5), whereas in the present experiment, separate ratings were given for each task (see Fig. 11). The multiple regression was done using an average of the precision hover and the landing Cooper Harper ratings.

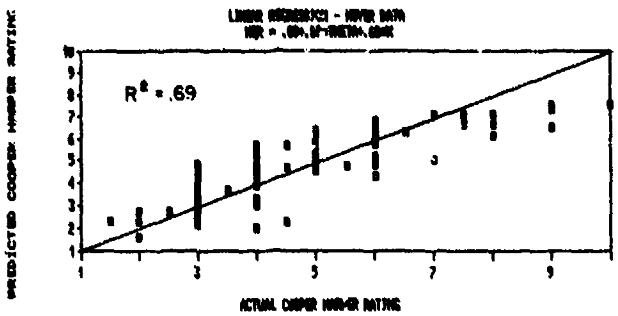


Figure 12. Linear Multiple Regression for all cases without ACAH (from both experiments) R^2 = 0.69, 89 observations, RMS error of Cooper-Harper Rating = 1.0

The data for Rate augmented and unaugmented configurations from both experiments are plotted on a grid of attitude VCR vs. horizontal translation VCR in Fig. 13. The dashed lines represent estimated Cooper-Harper handling qualities ratings (HQR) from the linear regression fit, and are seen to represent a reasonable (albeit conservative) separation between the pilot rating data. The data is separated at the 3.5 and 5.5 values of handling quality rating on the basis that the 3.5 line represents the classical Level 1/2 boundary in Mil-F-8785C (Ref. 10). The 5.5 line is based on the results shown in Fig. 13b, and the 6.5 line is Level 2/3 boundary in Mil-F-8785C.

The results shown in Fig. 13b indicate that the region defined by handling quality ratings of 3.5 to 5.5 for the baseline Rate Response-Types is mostly Level 1 for ACAH. All of the exceptions are barely Level 2 (rating of 4) and occurred in gusty wind conditions. As the visual conditions degrade beyond the line defined for HQR (Rate) = 5.5, the ACAH augmentation is seen to be ineffective as a means for maintaining Level 1 handling qualities. The 5.5 line is therefore a natural upper limit for a criterion which allows ACAH to compensate for a degraded visual cue environment.

IV. DEVELOPMENT OF CRITERION

Figure 13a suggests that the region below the HQR (Rate) = 3.5 line does not require additional stabilization, while Fig. 13b indicates that the region between that line and the HQR (Rate) = 5.5 line is Level 1 when ACAH augmentation is employed. A criterion suggested by these regions, with the following modifications, is given in Fig. 14.

- The regions have been modified to disallow extreme differences between attitude and translation VCR ratings as a means of compliance. This is to prevent, for example, a display with excellent attitude cues and poor translation cues from meeting the criterion.

- The region above the HQR (Rate) = 6.5 has been disallowed on the basis that it is unlikely that any augmentation can make up for such a major deficiency in visual cueing.

The regions established in Fig. 13 have been defined in term of four levels of usable cue environment (UCE) in Fig. 14. Each UCE level is utilized to set a requirement for a minimum Response-Type in Table 1. (The minimum response-types for the pitch and roll axes are shown in parenthesis in Fig. 14, below the UCE label). The justifiction for requiring Rate and RCAH for UCE=1, and ACAH for UCE=2, (Table 1 and Fig. 14) is based on the experimental data in Fig. 13. The justification for adding position hold in the UCE=3 region, is based on recent simulation data (not yet published) which showed that Level 1 ratings were possible with position hold, even when the pilot was preoccupied with other tasks in a very high workload environment. In addition, the simulator visual display (NASA Ames VMS) had a UCE of 2 (based on Fig. 7).

The Table 1 requirements for the yaw and height axis stabilization for UCE = 1 and 2 are based on what was used on the Bell 205 during the night vision goggle experiment. The Table 1 requirement for heading hold and altitude hold for UCE=3, is not supported by data at this time.

Application of the Criterion

The UCE ratings used in Table 1 must be obtained experimentally, using the VCR scale in Fig. 1, and conversion to UCE in Fig. 14. The process of obtaining the VCR ratings consists of an experimental evaluation of the proposed vision aids and displays, and must be conducted under certain specified conditions.

- The test aircraft must have a Rate or RCAH Response-Type. Additional stabilization would obviate the need for the cues that the display is being tested for.

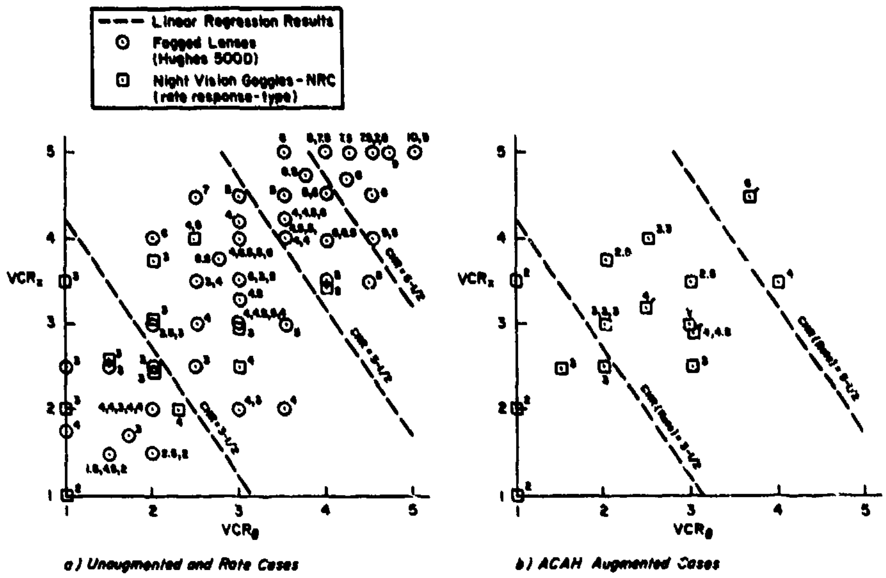


Figure 13. Correlation of Cooper-Harper Handling Qualities (HQR) and Visual Cue Ratings (VCR)

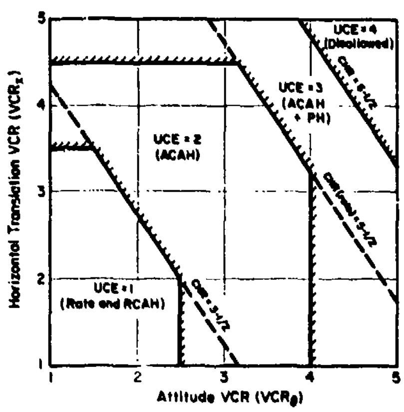


Figure 14. Definition of Usable Cue Environments and Minimum Allowable Response-Types

TABLE 1. REQUIRED UPGRADED RESPONSE-TYPE IN THE PRESENCE OF DEGRADED UCE -- NEAR-EARTH OPERATIONS

AXIS OF CONTROL	RESPONSE-TYPE SPECIFIED FOR UCE=1	UPGRADED RESPONSE-TYPE IN THE PRESENCE OF DEGRADED UCE	
		UCE=2	UCE=3
Pitch and Roll	Rate	ACAH	ACAH + PH
	ACAH	ACAH	ACAH + PH
Yaw	Rate	Rate	RCDH
Height	Rate	Rate	Rate + RCHH

NOTES:

- ACAH -- Attitude Command/Attitude Hold
- RCDH -- Rate Command/Directional Hold
- PH -- Position Hold or "hover hold"
- RCHH -- Altitude Rate Command with Altitude Hold

- The test aircraft must be Level 1 in good visibility (i.e. the average handling qualities ratings must be 3-1/2 or better).
- At least 3 evaluation pilots must be used and their results averaged (hence the need for a linear VCR rating scale).
- The tests should be conducted in calm air.
- The tests should include precision hover, precision vertical landing, hover turns about a point, quickstops, and bobup and bobdown.
- The standard deviation of the VCRs should be less that 0.75 or additional pilots should be employed, or the procuring activity may designate the required upgrade. The caveat is included to allow the removal of an anomalous rating which may occur, for example due to a pilot's preconceived notion regarding a particular display.

Note that it is not necessary, or even desirable, to test the display in the prototype aircraft. The handling qualitites of such an aircraft are rarely well known, the display may be ready for testing before the test aircraft, and it is not desirable to tie up a prototype test aircraft to evaluate displays.

V. REQUIREMENTS FOR IMPROVED VISUAL DISPLAYS

As noted above, the use of control augmentation to offset a degradation in visual cueing represents a compromise in which agility, and aggressiveness are sacrificed. A better, albeit not currently attainable, solution is to provide a display with adequate field-of-view and range for positional awareness, and microtexture for control and stabilization.

The implication of the results presented herein is that microtexture is an important cue which must be quantified in order to develop meaningful display requirements. Such quantification would be couched in terms of the modulation transfer function (MTF) which characterizes microtexture in terms of spatial frequency (Ω), and the modulation of the image (Ref. 11). The modulation of the image is measured as the difference in intensities between the peaks and the troughs across the spectrum in the visual field. Hence the contrast of the microtexture can be quantified in terms of the depth of modulation.

The resulting display requirements might appear as shown in Fig. 15. The upper limit of the required modulation depth is based on the maximum capability of the human eye as measured by Van Ness and Bouman (Ref. 12). The lower limit is an estimate since data are not available. Similarly, the desired range of spatial frequencies is a rough estimate, centered about the frequency of the cracks in the lakebed (at a range of 20 ft. from the pilot's eye) available on the test course in the Ref. 4 experiment (see Fig. 5b). The lower curve in Fig. 15 "explains" how some pilots achieve 20/20 visual acuity with the lenses fogged. That is, there was probably sufficient depth of modulation at a spatial frequency of one arc-minute to distinguish the letters on the eyechart, but not to acquire the information required for precision hover maneuvers.

Lacking precise, quantitative measures, such as the modulation transfer function, the VCR scale (Fig. 1) has been derived to measure the usable cue environment in terms of the ability to maneuver aggressively. The results of these experiments indicate that the scale is reasonably successful, and on that basis, is used to define the useable cue environment associated with a given display or vision aid. It was found that strict adherence to making assessments based on the level of achievable aggressiveness, as opposed to the pilot's qualitative evaluation of the available visual cues, is necessary when using the scale.

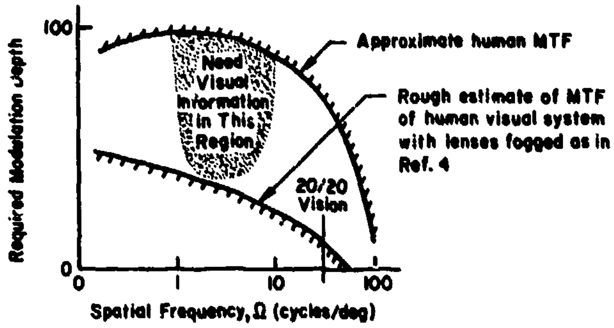


Figure 15. Proposed Generic Form of the Required Region of Visual Information for Hover

VI. CONCLUSIONS

The following conclusions summarize the development of the criterion developed in this paper.

- There is considerable evidence that microtexture is a primary cue for control and stabilization in hover and low speed flight.

- Field-of-view is of secondary importance to microtexture for control and stabilization, although it may be highly significant for positional awareness. Experimental data shows that the 30 deg field-of-view available on the PVS-6 night vision goggles is adequate for control and stabilization.

- It is possible to estimate the effectiveness of a display in terms of the visual cue rating (VCR) scale, and the resulting usable cue environment (UCE). These ratings may be used to assess the need for additional stability augmentation via the criterion developed herein.

- It is possible to makeup for losses in visual cues with attitude agumentation.

- The use of attitude augmentation to makeup for display deficiencies (i.e., insufficient microtexture) usually results in a loss of agility. Therefore, it is more desirable to improve the visual cueing than to makeup for a loss in such cueing via augmentation.

- It would be desirable to develop a more quantitative metric to evaluate displays. For example, the requirement could be stated in terms of an acceptable region on a grid of depth of modulation vs. spatial frequency. Research needs to be accomplished to determine this region.

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MBB SIMULATION FACILITIES APPLIED FOR ROTORCRAFT RESEARCH

by

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SUMMARY

The attitude in the rotorcraft industry towards simulator applications during helicopter development work has been changing fundamentally in the past decade. More and more demanding design requirements for future helicopter systems, and the efforts to replace current helicopter flying qualities criteria by new specifications allowing for mission-oriented handling qualities, have significantly upgraded the role of simulation in the design process. Rapid advances in recent years in the level of sophistication and fidelity of modern ground-based simulators have been major contributions to this change.

Man-in-the-loop real-time simulation for the design and system investigation of advanced aircraft is applied in the MBB fixed-base flight simulation center, both for fixed-wing and rotorcraft simulation. The simulation facility, which presently comprises a basic three channel CGI visual system with collimator projection, is being expanded by installation of a more powerful 8 channel CGI system with six light valve projectors providing an extended field-of-view on the inner surface of a sperical dome. The visual system takes into account various helicopter-specific requirements such as a topographical data base with highly detailed low level visual flight cues for NOE missions, sensor image presentation (e.g. IR, LLLTV) with selectable fields-of-view, two independently controllable viewpoints for two crewmembers, and the selection of different moving objects superimposed on the scene. These features make it particularly suitable to analyze mission requirements, and to investigate configuration design and system layout.

The increasing pilot workload and environmental conditions of modern military helicopters necessitate detailed assessment of cockpit designs with regard to controls and displays. The optimization must be performed at the initialization of the concept to ensure a harmonic design with minimum pilot workload. The status of the MBB flight simulation facility is examined more closely in the lecture in light of these challenging requirements. Projections of the further development of the simulation center components are depicted, defining priorities as they are seen by the company. Consequences are outlined that have to be considered when organizing program logistics in order to guarantee the success of simulation investigations.

1. INTRODUCTION

In the past decade, requirements for the design and development of future rotorcraft have experienced a tremendous change. New operational mission requirements evolve from the expanding role of the helicopter and from the changing battlefield scenario. New cockpit-related technologies, like

- cockpit displays
- visual aids (sensors)
- improved cockpit control devices
- all-embracing on-board data handling

make the next generation helicopters technically even more complex than they were before. Recent progress in rotorcraft technology has evoked a variety of sophisticated concepts projected to overcome the restrictions of today's rotorcraft configurations. The most promising are

- hingeless rotor helicopters
- compound helicopters
- tilt-rotor aircraft
- X-wing aircraft
- ABC helicopters.

All these concepts have created a lot of new problems which have to be solved in future work. This has significantly upgraded the role of simulation, which is now more and more used as an indispensible and valuable engineering tool in the rotorcraft design process [1] . Rapid advances in recent years in the level of sophistication and fidelity of modern ground-based simulators were major contributions to this trend. On the other hand, the need to apply simulation methods in advanced helicopter development clearly requires advanced simulation methods themselves. High flexibility in rotorcraft simulation equipment is needed to allow for feasibility and trade-off studies and for frequent changes resulting from design and development iterations.

Flight simulation is used today in a vast field of applications, ranging from fundamental research in scientific institutions to simulations performed by the final customer to do operational flight planning or pilot training (Fig. 1.1). In the rotorcraft industry, flight simulation is applied during various levels of the design and development process. The status of the MBB flight simulation facility will be examined more closely in the context of the challenging requirements listed above. Projections of the further development of the simulation center components will be depicted, defining priorities as they are seen by the company. Consequences will be outlined that have to be considered when organizing program logistics in order to guarantee the success of the simulation investigations.

2. ARCHITECTURE OF THE MBB FLIGHT SIMULATION

Flight simulation is used to reproduce the characteristics of any given aircraft by means of technical equipment at places other than the original surroundings. In this lecture, we will concentrate only on ground-based simulation methods, leaving out the special capabilities of in-flight simulators.

Modern ground-based simulators usually comprise simulation of motion, visual and aural cues to provide the most realistic stimulation of the pilot's sensoral receptors. A representative simulation cockpit, providing adequate space and field-of-view, cockpit displays, operation, and control elements to the flight crew, is commonly used. Several other components are needed to accomplish the simulation process, so today's flight simulators have grown to be comprehensive and complex simulation centers or labo-atories, employing their own staff. In light of the continuous evolution in aircraft engineering, such flight simulation facilities are, and have to be, constantly changing and dynamic entities [2] . Although this challenge is at present more often realized in the fixed-wing industry, the rotorcraft industry obviously must begin to follow this evolutionary organizational quantum leap as well.

At MBB Munich, both fixed-wing and rotorcraft branches use the same simulation laboratory. Thus they simultaneously take advantage of the technological developments which are continuously being incorporated into the simulation facility. For better understanding of the application of flight simulation in engineering tasks, we will start with a discussion of the different components of the MBB Flight Simulation Center (Fig.2.1).

2.1 HELICOPTER SIMULATION MODEL / SIMULATION COMPUTER

Briefly speaking, technical simulations are based on mathematical descriptions of physical and mechanical processes by means of equations of motion. All force and moment producing systems, subsystems, and components of the aircraft have to be considered and represented in the mathematical simulation model (Fig.2.2). In the simulation computer this universal mathematical description of the dynamic behaviour of the aircraft is organized in a generic program structure which may be used for any given type of aircraft without adaptation. By use of different data sets for geometries, masses, inertias, and aerodynamic characteristics, the program may be easily and quickly matched to any aircraft to be simulated.

This has been prepared at MBB, both for fixed-wing aircraft and for rotorcraft. The needs of piloted simulations are met by real-time processing of the mathematical model equations on a simulation computer that is capable of parallel processing. As a typical example, all rotor blade aerodynamic and dynamic computations are calculated in parallel (Fig. 2.3). Due to its modular structure, additional aircraft components may be modelled and attached to the generic program without great effort. Thus flexibility required to allow for the addition of any new aircraft component is ensured.

Simulation software development will continue to be a never-ending task. On the one hand, progress in rotorcraft simulation modelling demands continuous adaptation of the program state-of-the-art. So the generic model has to be furnished with new routines, e.g. for Automatic Flight Control Systems, weapon simulation, or models for the simulation of new concepts like tilt rotor or X-wing.

On the other hand, the tremendous increase of airborne software promises new cockpit technologies like artificial intelligence in the cockpit. Software check-out and validation is urgently needed as well as adequate software development systems which make the simulation process directly accessible to the design engineer.

As in many other companies, off-line engineering simulations at MBB Munich are usually run on computer systems other than those used for real-time simulation. This is due to the different requirements of both simulation methods. Therefore, in order to guarantee the compatibility of simulation software and data sets on all systems, well- defined software concepts including the consideration of software portability are imperative. And, most important but also most neglected, excellent program documentation is a factor of software development which shows its payoff only in long-term application.

In the U.S., these demands led to the Department of Defense (DOD) mandate for the future use of the ADA computer language for all Mission Critical Programs [3] . An implementation program which aims for the use of ADA as the programming language for simulators by 1987 has been issued, and US Army and Navy have announced plans for internal crew training to meet the ADA requirement. However, as the development risk of changing an existing comprehensive software package to a new program language is considered to be too high, no effort is at present being taken at MBB to implement any new higher-order language.

2.2 SIMULATION COCKPITS

In order to provide the flight crew with a representative cockpit environment with relevant space and field-of-view proportions, cockpit displays, operation, and control elements, a number of interchangeable flight simulation cockpits is usually available at flight simulation centers. They are equipped with the appropriate instrumentation and connections and may be prepared and made available for real-time simulation runs within a short time. For basic investigations and preliminary studies in early project phases it is oftentimes sufficient to use a standard cockpit assembly. Therefore, a Bo105 fuselage (Fig.2.4) was the first simulation cockpit which was used intensively at MBB for simulation model validation and fundamental investigations [4] . This is an original prototype aircraft from flight test with primary flight controls and conventional instrument panel, equipped and modified for simulation purposes. However, the main and tail rotor, the tail boom, and the engines have been removed for better handling in the facility.

For detailed investigations, say in the project development phase, construction of a representative simulation cockpit is indispensible. In Fig.2.5, the simulation cockpit for the next generation anti-tank helicopter, PAH-2, is presented. This is a special purpose mockup which had to be fabricated to achieve the characteristic narrow width and the typical limitations in the field-of-view of a tandem seat tactical helicopter in simulation. At present, this cockpit is fitted with conventional flight instruments, but in the next development stage up-to-date multi-function displays and visionics will be included. The simulation tool has already been used in the recent feasibility and pre-definition phase of this project.

Many cockpit hardware components are not completely specified in the early project phases, nor necessarily available. In order to make simulations feasible in these development stages, it is frequently necessary to install preliminary mechanizations which may differ widely from the later hardware implementations. The true characteristics of those components may then be met by adopting the software models appropriately [5] .

In today's flight simulation centers, the trend is pointing towards the operation of standardized simulation cockpit platforms furnished with cockpit controls with variable control loadings (electromechanical or hydraulic control forces simulation) and interchangeable flight instrumentation. Modern digital cockpit display techniques which are used in the new generation aircraft to achieve flexibility in flight status presentation, contribute to the progress in simulation technology development. Space and field-of-view proportions may be adjusted by mounting mockup constructions and blinds so as to allow for quick and low-priced configuration changes. In Fig.2.6 an example of such a cockpit mockup is shown which is presently used at MBB for basic studies for cockpit layout, investigations for display and operation concepts, and the development of new cockpit control systems.

2.3 MOTION / ACCELERATIONS SIMULATION

The pilot's evaluation of flying qualities is predominantly based on the stimulation of his various sensory receptors. For that reason, mature flight simulation centers comprise motional and visual information with additional cues, such as acoustic signals. For a long time, requirements for an adequate simulation of accelerations and visual scene display formed the most relevant handicaps for simulating rotorcraft dynamics in real time.

Considering the new generation of high performance computers with their outstanding computing capacities, these restrictions lose their importance more and more. However, the immense investment costs oftentimes force the decision to install either a motion or a visual simulation system, and, unfortunately, there is still no definite answer as to which method is the most important one [6] .

Decision has been taken at the MBB flight simulation center to put first priority on a good visual system, to the disadvantage of motion simulation. So the facility is equipped with no more than a one-axis vertical acceleration hydraulic system, allowing for the simulation of z-loads, like vibrations, turbulence, or buffeting, up to ±2 g. Further performance improvement will be achieved in the very near future by providing an off-the-shelf g-cueing suit for the simulation pilot.

2.4 COMPUTER GENERATED IMAGERY (CGI)

Major requirements for visual simulation of out-of-the-window real-world scenes derive from two different factors: Firstly, the overwhelming performance capabilities of the human eye, and secondly, the operational flight regimes and missions of the respective aircraft that is to be simulated [1] . We will leave out here the discussion of the fundamentals of the human visual perception, as they have been addressed in several excellent publications, like e.g. [7,8].

When we consider the helicopter-specific flight regime, visual scene simulation in particular requires a high-resolution wide-angle field-of-view (FOV). Helicopters typically fly low and slow, and "Nap-of-the-Earth" (NOE) has become a well-known term to describe tactical point-to-point flying and hover operations in close ground proximity (Fig.2.7). Moreover, steep take-off and landing procedures also constitute typical helicopter flight profiles. The environment for the pilots flying these missions is rich in detail, and terrain features, as well as the visibility factors of weather and darkness, are elements of the environment that may significantly affect the helicopter pilot's task [9] .

Other vital demands for a large FOV are posed by the outstanding ability of helicopters to yaw rapidly. Based on pilot's minimum preview times for obstacle avoidance, and dependent on the actual yaw rate as well as on the roll attitude, the visual scene must cover obstacles that are some degrees ahead in the projected flight path azimuth. Piloted simulation of quick stop maneuvers for a fast transition from forward to hover flight or the recovery from autorotation is almost unacceptable when visual cues get completely lost at large nose-up pitch attitudes because of the limitations in the vertical FOV.

Since the early 70s, the technique to generate computer imagery has been constantly improved. This technology now seems to have reached a point where it can supply all the desired details, the limiting factor for a common use of those systems now as before being the high final costs [10] . In computer imagery generation systems (CIG) the information about the physical dimensions of the scene surrounding the aircraft, such as terrain, buildings, trees, and diverse artifacts, is stored in a computer's memory. The scene is continuously processed to make the pilot feel that he flies through the scenario. Therefore, geometric transformation algorithms are used which take into account the aircraft's three-dimensional position in space, its attitudes and velocities. Corresponding to the pilot's momentary field-of-view, the out-of-the-window sight is generated and displayed in real-time in front of the simulation cockpit.

The geographic area to be modelled is practically limited only by the memory capacity and the speed of the computer. The vast majority of the scene is invisible at any one time, including what is behind the aircraft, what is too far ahead to be visible or what is hidden behind more prominent features in the line of sight [11] . Nevertheless, the level of detail of the imagery is limited, as the display has to be generated dynamically in real-time, bringing into discussion such terms as 'refresh rate' and 'update rate' of the picture.

In Fig. 2.8, an airfield is shown as it is presently implemented in the MBB COMPU-SCENE II visual system. This is a General Electric three-channel system with Color-TV type projection, where the three optical axes meet in the pilot's eyepoint, defining a field-of-view (FOV) of 26° vertically by 106° horizontally. Collimator lenses are installed between the pilot and the display screens in order to make the light rays parallel as if they were reflected from infinity. The system allows the generation of up to 8.000 edges per scene. Various visibility conditions, such as haze, fog, clouds, and day-, dusk-, and night-time may be simulated. The image is processed with a refresh rate of 30 Hz, rewriting the scene every 33 milliseconds. Fig. 2.9 gives an impression of the Bo105 simulation cockpit, integrated in the projection system assembly.

At present, MBB Munich is improving its flight simulation center with the next-generation COMPU-SCENE IV dome projection system (Fig.2.10). The new system will be ready for use in early 1987, bringing a remarkable improvement of the visual scene display characteristics:

Field-of-View: Compared to the FOV of the former system, a considerable extension of the overall FOV will be achieved by using 6 independent projection channels (Fig. 2.11). It will extend 115° vertically by 140° horizontally, which will meet the requirements of helicopters much better. Projection is performed from the outside onto the reflective inner surface of a sphere of 10 m diameter (Fig. 2.12). By adding two more projection systems there is a future optional FOV expansion to 150° by 300°.

Gaming Area: U.S. Defence Mapping Agency (DMA) geographic information is used to generate a landscape model of 200 by 200 nautical miles. This environment will include areas of farmland, rolling terrain, desert and ocean regions (Fig. 2.13), as well as airfields, buildings, trees, and other special features (Fig. 2.14). An additional data base designed for helicopter operation will be included. This data base will be of much higher scene density to support the lower and slower flight regime of rotorcraft vehicles. It will contain the trees, bushes, and ground clutter necessary to meet the stringent visual/sensor requirements of below-treetop height flight maneuvers.

__Texturing:__ Extensive use will be made of cell texture maps which will be developed from photographs, models and the use of noise patterns. A library of generic surfaces like scrubland, desert, beach, farmland, forest, roads, generic sky/cloud patterns (Fig.2.15), and special highly detailed shapes of trees (showing even single leaves), buildings, and rocky clutter will be available. Other effects will include curved surface shading in order to guarantee a good three-dimensional perspective, sun illumination as a function of time-of-day, and weather influences.

__Moving Objects:__ Moving models of various aircraft (Fig. 2.15), tanks, trucks, and trains will be modelled to be inserted at any desired position in the scene. Independent motion may be supplied in the simulation program to make those objects move around the scene. Disintegration of certain targets will be available to simulate the effects of explosions and weapon impact.

__Sensor Option:__ The image generator will provide the capability of sensor simulation through dynamic change of color tables and field-of-view in order to provide typical infra-red (IR) or low light level TV (LLLTV) sensor images (Fig. 2.16). Those views are then to be displayed on the respective displays in the simulation cockpit.

__Independent Viewpoints:__ The Image Generator will be capable of providing the visual scene for two viewpoints with each viewpoint having independent motion. Either viewpoint will be assignable to any display channel and capable of displaying either sensor and/or out-of-the-window visual scenes within the active data base.

Based on the features of the COMPU-SCENE IV, it can be seen that the visual simulation state-of-the-art is now obviously approaching a near photographic appearance [11] .

2.5 PERIPHERAL SYSTEMS

In the MBB flight simulation facility, the CGI projection dome and the simulation cockpit are in a room by themselves, and as are the simulation and the visual system computers (Fig. 2.1). There is a separate simulation control room which contains the associated computers, displays, and recording devices for supervising, monitoring, and recording the simulation flights (Fig. 2.17). From the control room, the simulation engineer-in-charge can also manipulate the simulation process so as to introduce certain malfunctions of diverse rotorcraft components (e.g. engine failure, tail rotor loss), change weather and visibility conditions on-line, move around target objects etc. The driver electronics for the digital cockpit displays, as well as the spectra generators for buffeting and turbulence inputs to the hydraulic g-load system are installed here. Cabin noise is another signal which will be generated here and fed into the simulation cockpit loudspeakers or pilot's earphones. And, last but not least, an intercom system to be installed will ensure effortless communication between the simulation engineer, the pilot, and all other personnel involved.

Data transfer between the simulation computer and the simulation cockpit is organized via an interface computer which controls all the necessary signal transformations (digital/analog, analog/digital, digital/synchro, etc.). From here the cockpit instruments are driven and supplied with power, and from here the measured cockpit control positions, switch positions, and all other pilot's hardware actions are digitized and supplied to the simulation computer. The interface computer is also capable of driving different data busses (ARINC, MIL) which now more and more form part of rotorcraft systems (such as fly-by-wire flight control systems).

In the near future, a separate program development and data analysis workstation will be added to the helicopter installation equipment in order support pre- and post-simulation off-line work (Fig. 2.18).

2.6 INTEGRATION RIG

The term integration rig describes hardware mountings or racks where existing hardware components of the aircraft are continuously being installed and tested in the course of the development process in a simulated environment. Frequently, the mathematical simulation model of the different rotorcraft components represents the complex characteristics of the system components only moderately well. They are therefore successively replaced by existing equipment until total system integration with signal flows representative of the final aircraft is achieved. This implementation includes the incorporation, as early as possible, of hardware components such as flight control systems, avionics, visionics, hydraulics, etc., with their true physics and dynamics in the simulation process. Or, vice versa, this serves to drive the installed and linked units with the completest possible set of dynamically coupled aircraft system parameters.

The controversy of coupling simulation facilities and integration rigs is still unresolved. A hardware integration rig itself is an exceedingly complex system, with major elements being the basic helicopter rig, the hardware equipment, the integration rig computer, and the associated integration software. More failures and problems have to be expected (and American experience acknowledges this) when we endeavor to connect this rig with an equally complex flight simulation laboratory. Despite the great effort being invested, these large-scale facilities may not be utilized effectively.

An obviously better solution is to run both assemblies independently, and to install only part of the equipment as 'hardware-in-the-loop' [12] . So in the PAH-2 development phase simulation it is planned to connect at least the AFCS laboratory sample via bus systems to the simulation cockpit and, therefore, to the simulation computer. The integration rig computer, on the other hand, will include a simplified rotorcraft simulation model, based on linearized equations of motion. This program will be used throughout the integration rig investigations for the generation of the flight state parameters set and for the stimulation of the sensors.

3. MBB HELICOPTER DESIGN AND DEVELOPMENT SIMULATION

3.1 SIMULATION MODEL VERIFICATION / VALIDATION

Once the generic simulation program has been established, a basic verification of the mathematical model is to be conducted to assure the further merits of the simulation application. This is best to be accomplished for an existing aircraft by comparison of simulated trimmed states and time histories with flight data, analysis of eigenvalues and eigenvectors, and by pilot evaluation. In order to apply the terms correctly, we will refer to the following definitions given in [13] :

<u>Verification:</u> The comparative assessment of the rotorcraft system as it is simulated by the most realistic available model(s). This is quantified through a comparison of the rotorcraft mathematical model(s) and the actual flight vehicle.

<u>Validation:</u> the degree to which transferability is demonstrated between all aspects of simulation performance and corresponding aspects of flight performance. Validation involves all aspects of the rotorcraft simulation, the mathematical model, and the motion, visual, and aural environments. The accuracy of the answers is critical if the simulation results are intended to quantify expected benefits prior to actual demonstration in flight.

Therefore, verification is a necessary but not a sufficient aspect of the future validation tasks. Verification work may usually be taken to be completed when the results show such good agreement as the example given in Fig. 3.1.

3.2 SIMULATION PREPARATION AND LOGISTICS

In the early program stages, simulation activities are usually characterized by trying as best as possible to simulate an aircraft which does not yet physically exist. As mentioned above, work is based on a generic simulation program which has already been used for other helicopter types, as well as on wind-tunnel measurements and subcontractor information (e.g. about the engine dynamics). In Fig. 3.2, an example is given of how simulation preparations are usually being performed as long as only a limited amount of the new system parameters is available; but a full set of data is required for overall flight envelope simulation. At the beginning of the design and development simulation, a great deal of the mathematical modelling of the helicopter has to be done, based on the engineer's experience, or by comparing the new helicopter with known characteristics of existing rotorcraft. This has to be done as long as no final specification of all components of the aircraft is available.

The results of the early simulations are therefore inevitably limited to the knowledge at the time of the investigation runs. There is no doubt that, with the cumulating results of the development process, simulation fidelity will constantly improve.

In order to guarantee the success of future simulation investigations, and to achieve a good lead time advantage, logistic preparations of the simulation tasks should take place as early as possible in the program. This includes management objectives scheduling and budgeting, as well as hardware, software, and data set preparation.

Simulation at its best is a highly efficient tool for the design and development engineer, provided it is used in a timely and effective manner. It is an indisputable benefit that the long-range items which have to be defined in the different program stages, may be more and more based on the predictions of the simulation studies results. However, it should be well-understood that simulation inevitably has to be prepared and budgeted well in advance of, or in the early part of the respective design and development phases. Thus, simulation will always be an important cost factor in the earlier development phases, because of the required hardware investment and the usually large number of investigations and trade-off studies that have to be performed then. And, moreover, it will show its pay-off usually at the end of the development process, when the administrators start to review the overall costs of the project.

3.3 APPLICATION OF FLIGHT SIMULATION

The broad range of application of flight simulation among research institutes, industry and customer has already been outlined in Fig.1.1. Referring to the different industrial phases of the rotorcraft design and development process, the role of simulation as it is understood at MBB will be illustrated in the following section. Although this is chiefly related to our company, many of the arguments are valid for any other flight simulation center.

Feasibility Study. Apart from the basic research studies carried out for fundamental research, market analysis, and acquisition, simulation is applied, as was mentioned before, very early in the course of the design process in order to investigate the feasibility of new flight vehicle or technology concepts. This is where the target specifications of the customers have to be analyzed and refined towards a requirements catalogue which will serve as basis for the draft request for proposal and the later development contract. Engineering simulation in this stage is oftentimes handled by means of mathematical models used to describe the characteristics of newly developed hardware. Typical tasks are preliminary investigations of the application of new technologies, such as Active Control Technology or Artificial Intelligence. Pilot evaluations of controllability of new systems in heavy duty conditions and flight mission profiles is another task of particular interest.

The advantage of using simulation in this early project stage is to make development risks calculable with relatively low costs, and to let the aircrew fly aircraft and system concepts that exist as yet only on paper. Expectations, be they too high on the customer's or on industry's side, and undesirable technical developments may thus be avoided in time.

As an example for a recent MBB project which is now in the Feasibility and Pre-Definition Stage (FPDS), the artist's view of the NH90 is given in Fig. 3.3. This is a medium size European helicopter which will be built in two versions, as the NATO Frigate Helicopter (NFH) and as the Tactical Transport Helicopter (TTH). Participating countries in the program are Germany, France, Italy, The Netherlands, and the United Kingdom. MBB preparations for the scheduled 40-50 hours of simulation time in this program (CPU time on the simulation computer) are currently at full blast.

Definition and Specification Phase. The simulation plays another important role during the Definition and Specification Phase, which is used to detail the requirements for the overall system and the subsystems. Just like modern fighter aircraft, today's military helicopters are increasingly complex weapon systems. The development of these flight vehicles must not concentrate only on the optimization of the basic system. It has, additionally, to aim towards the best possible harmonization of the various system components, such as flight controls, display and visual aids, weapon systems, and the flight crew. It is therefore imperative to simulate the overall system with all its subcomponents and interrelations. Concept studies have to be carried out to investigate the acceptability and operability, and to demonstrate the system performance.

Development Phase. In the development phase, the primary task of the simulation is to state whether or not the pilot is able to fulfill the specified missions with the projected aircraft system, as well as to state to which extent the necessary system modifications will influence the pilot-helicopter system. This classical simulation task is more and more becoming a standard process in rotorcraft development, too, as the potential of simulation is addressed here at its best. Parameter variations of the new system may be investigated without effort, thus providing the high flexibility needed for the trade-off studies. Besides that, the pilot may become acquainted with the dynamic characteristics of the new flight vehicle long before the flight test phase begins. The easy way to reproduce simulation runs under the same circumstances facilitate simulation flight test series to be performed with different pilots. Moreover, critical system failures (tail rotor loss, engine failure) may be investigated without risk. The potential to conduct closed-loop hardware integration testing in the course of the development phase has been mentioned before.

3.4 SIMULATION TEST RESULTS EVALUATION

The post-simulation analysis is usually carried out by the requesting department, with the assistance of the simulation engineers, if necessary. It is a common experience that a good interface definition between the cooperating departments is of great help for running the system effectively. So the department that is requiring simulation investigations is asked to fill in a precise simulation test form, defining the tasks to be performed as well as the input/output parameters desired. This should be handled strictly comparable to flight tests, where all the preparations have to be concluded before the test aircraft gets airborne, and *rien ne vas plus*.

An adequate evaluation of the real-time simulation results has to be conducted continuously throughout the design and development process. This can be done by:

- a comparison with the dynamics of existing helicopters with comparable characteristics (flight test results);
- a comparison with the results of off-line calculations, based on non-linearized equations or more sophisticated simulation models which cannot be executed in real-time;
- review of pilot's comments on the dynamic behaviour of the aircraft.

The need for off-line processing of the recorded data is self- evident. If possible, the same data analysis and graphic routines should be used, both in flight test and in simulation test in order to provide for compatibility of the results.

Efforts to replace current helicopter flying qualities criteria are going to culminate in the near future into a proposal for a new airworthiness design standard. Although primarily related to the U.S. LHX project, this is assumed to be the long overdue replacement for the old MIL-H-8501 A. Manifold use will be made of comparisons between flight test and simulation test results, thus up-grading the role of simulation in the design process effectively. And there is also no doubt that ground-based simulation will play an increasingly important role during certification procedures, and will incrementally be accepted as a valuable means of demonstrating compliance with existing certification regulations. As a recent example, the Federal Aviation Administration (FAA) has certified two helicopters (Bell 214 ST, Sikorsky S76 B). For the first time, autorotational landings have been accepted that were demonstrated solely by computer simulation.

4. CONCLUSION AND PROJECTIONS

Flight simulation has evolved to be a primary and most efficient tool for the rotorcraft design and development engineer, and the flight simulation methods and tools themselves are continuously improving. The potential of the simulation technique appears not yet to be exhausted, especially in rotorcraft technology. However, the task of running today's flight simulation laboratories more and more includes the challenge to plan for tomorrow's facilities in time. This turns out to be an ambitious and ever-lasting enterprise for the simulation engineers, trying to incorporate a dynamically growing computer technology into short and long-term planning of business managers who are more familiar with thinking in effectivity terms.

It may be allowed to finish with a modification of a question that was posed on an earlier occasion [14] to describe the future challenge of simulation work:
The simulation engineers will continuously have to give an answer to the question "How do we manage flight simulation to keep up with the tremendous advances in computer technology and flight engineering?"

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SCIENTIFIC RESEARCH INSTITUTES	INDUSTRY	CUSTOMER/ AUTHORITIES/ USER
FUNDAMENTAL RESEARCH	FUNDAMENTAL RESEARCH	
REQUIREMENTS	MARKET ANALYSIS	REQUIREMENTS
	FEASIBILITY	
	DEF./SPECIFICATION	CERTIFICATION
	DEVELOPMENT	
ACCIDENT ANALYSIS	FLIGHT TEST	OPERATIONAL FLIGHT PLANNING
	BRIEFING	TRAINING

Fig. 1.1: Range of A ication of Flight Simulation

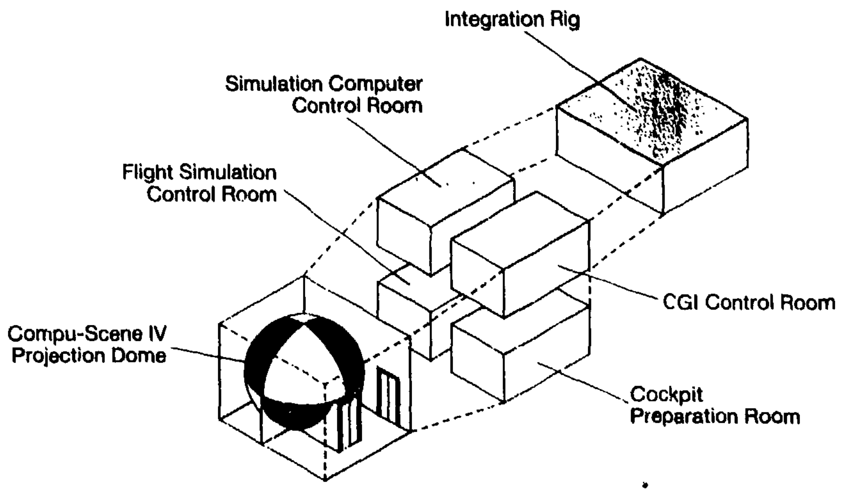

Fig. 2.1: MBB Flight Simulation Center General View

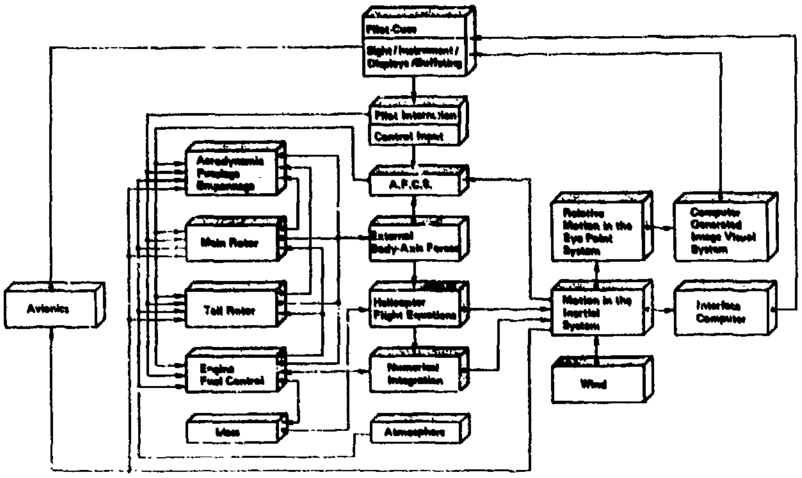

Fig. 2.2: Block Diagram of Generic Rotorcraft Simulation Model

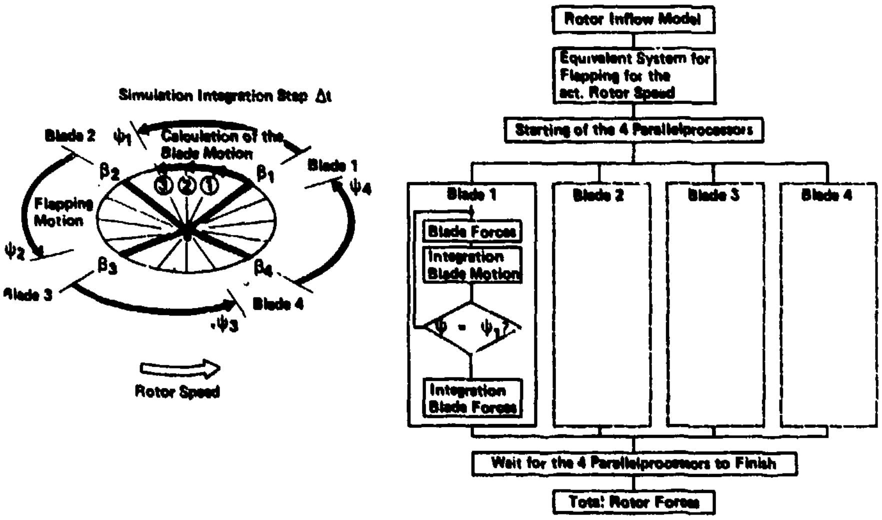

Fig. 2.3: Calculation of Rotor Blade Dynamics

Fig. 2.4: Bo105 Simulation Cockpit

Fig. 2.5: PAH2 Simulation Cockpit Mockup

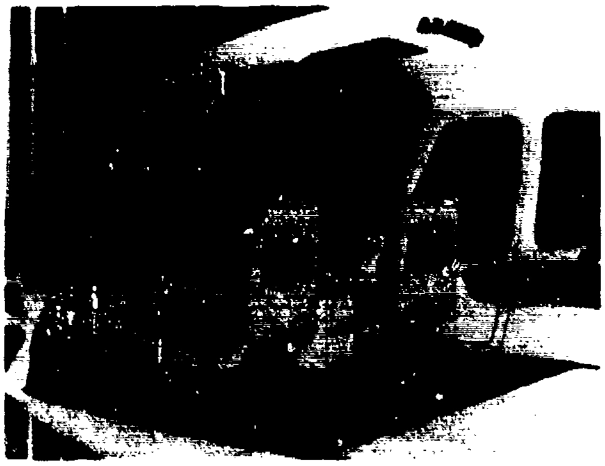

Fig. 2.6: Future Light Helicopter Simulation Cockpit Mockup

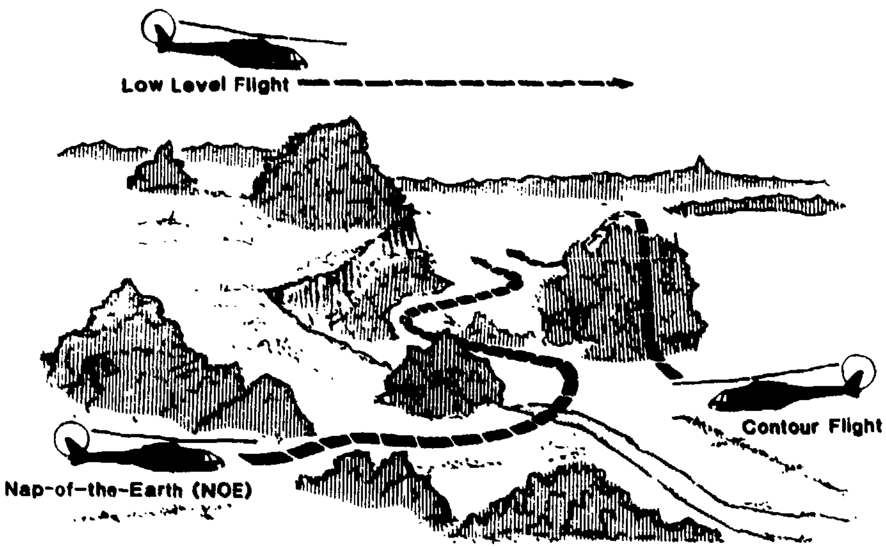

Fig. 2.7: Typical helicopter mission flight profiles.

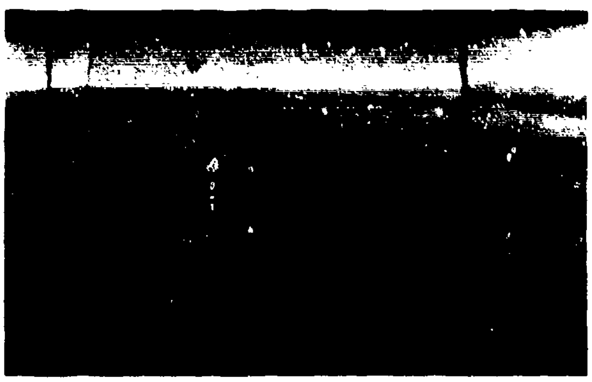

Fig. 2.8: Computer Generated Airfield View

Fig. 2.9: MBB CGI System COMPU-SCENE II

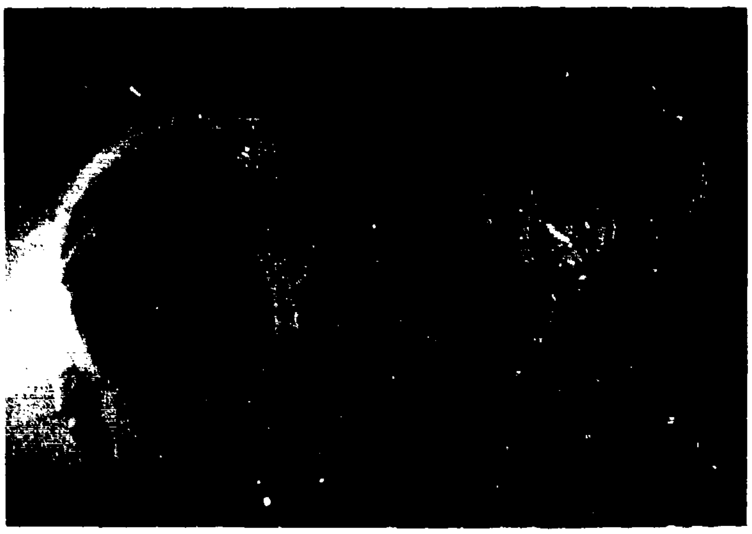

Fig. 2.10: MBB CGI System COMPU-SCENE IV

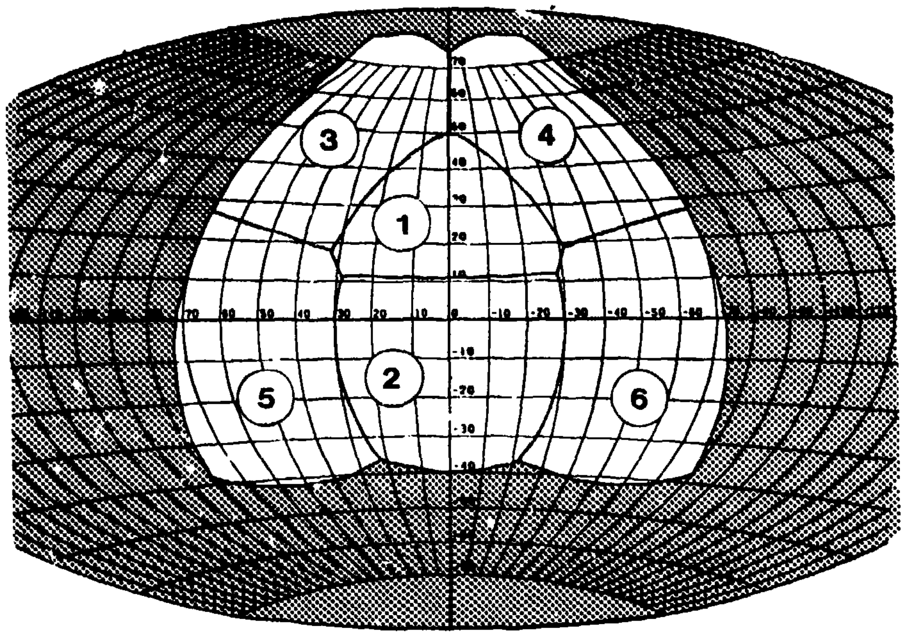

Fig. 2.11: COMPU-SCENE IV Six-Channel Field-of-View Details

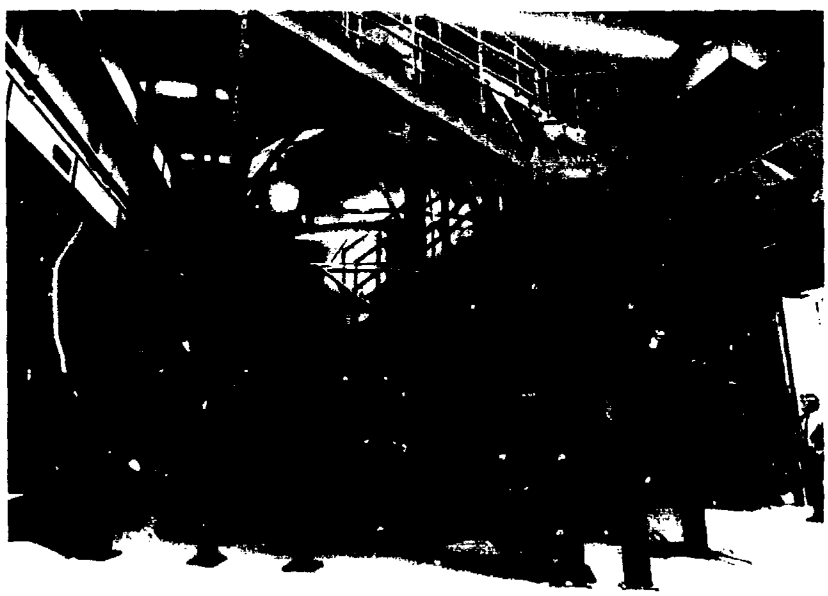

Fig. 2.12: MBB COMPU-SCENE IV Projection Dome

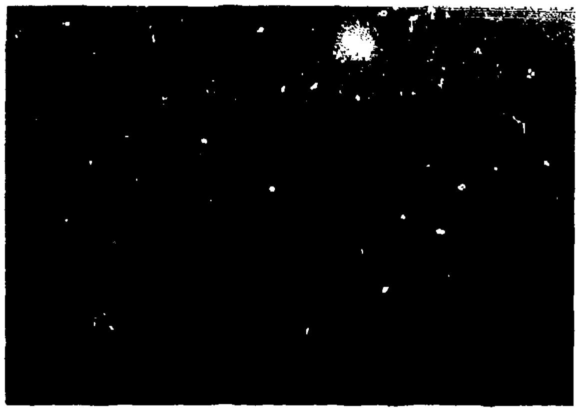

Fig. 2.13: New CGI Generation Rolling Terrain

Fig. 2.14: New CGI Generation High Detailed Scene

Fig. 2.15: New CGI Generation High Altitude Flight

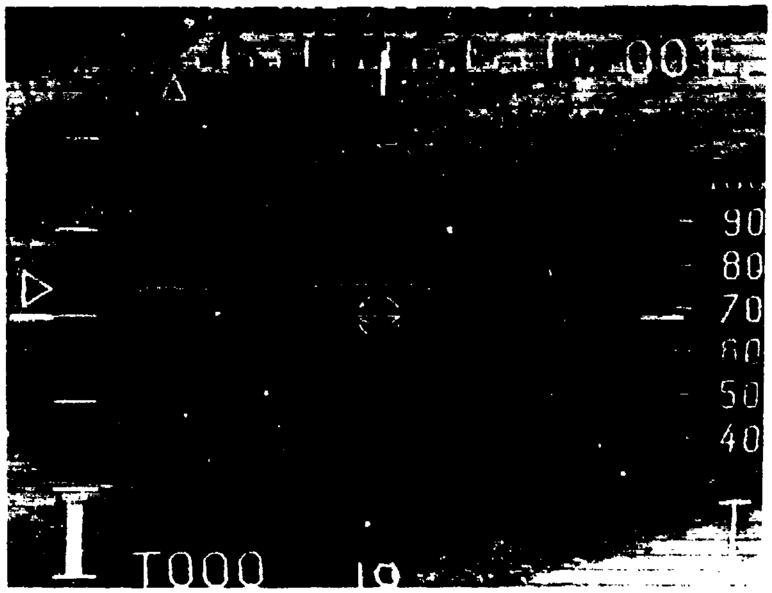

Fig. 2.16: Typical Infra-Red Sensor (FLIR) Image

Fig. 2.17: MBB Flight Simulation Control Room

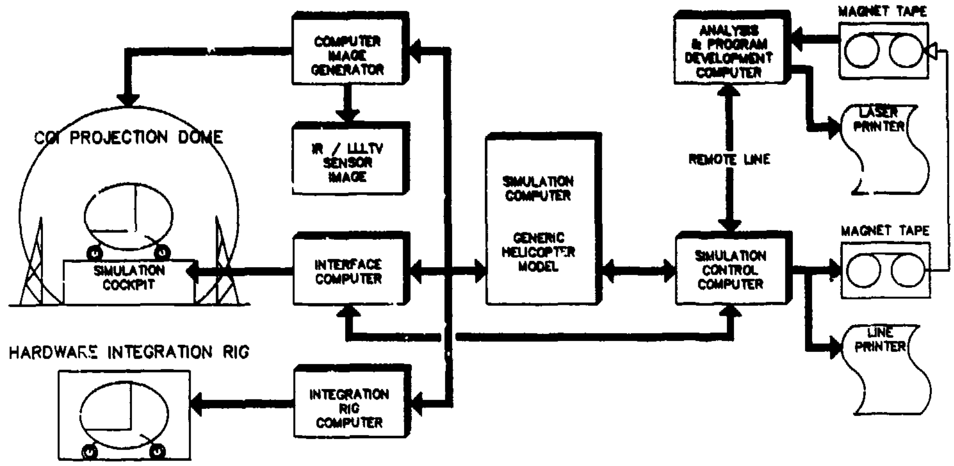

Fig. 2.18: Future MBB Flight Simulation Center Development

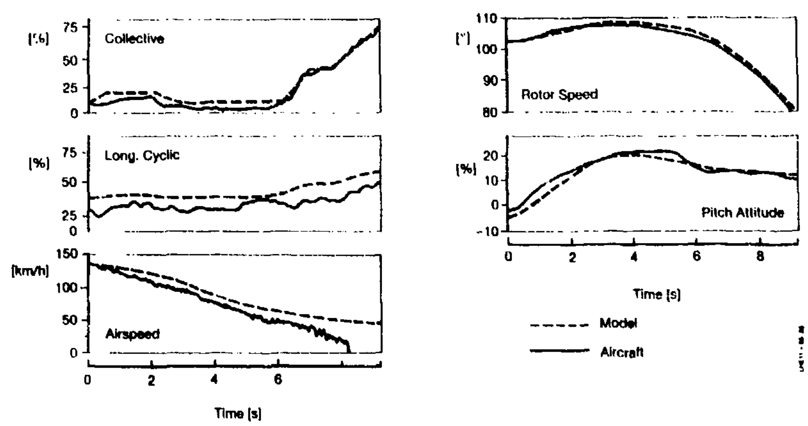

Fig. 3.1: Comparison of Time Histories from Simulation and Flight Test

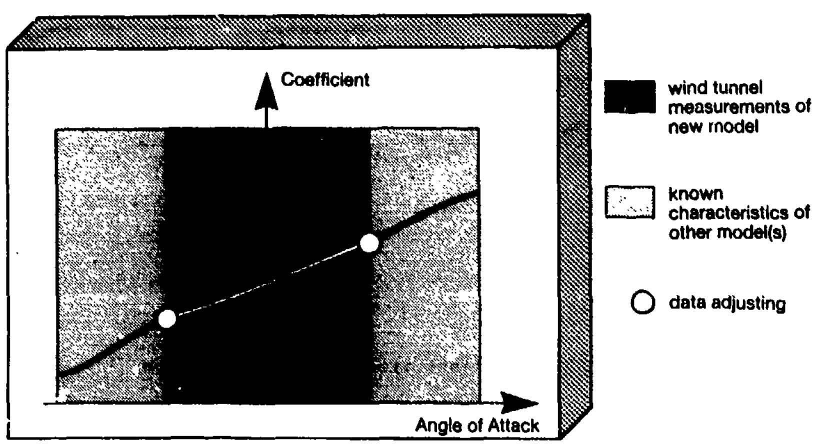

Fig. 3.2: Example of Data Preparation for Projected Aircraft

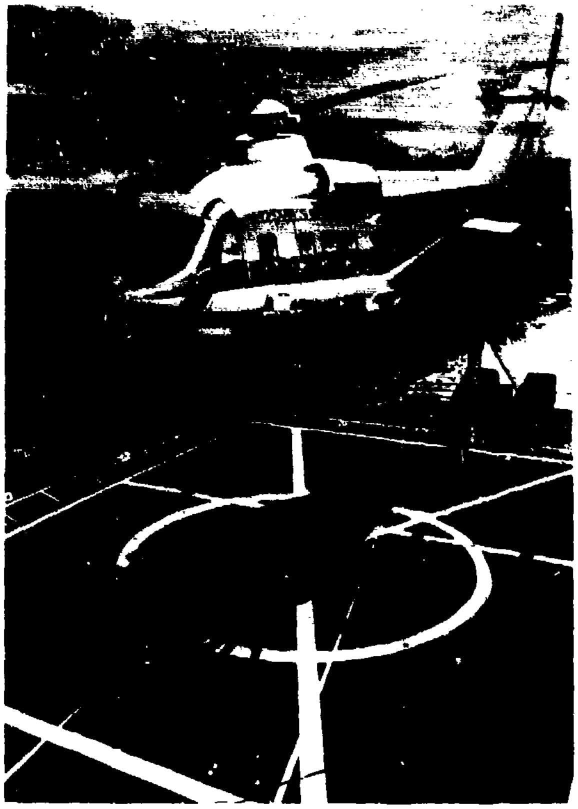

Fig. 3.3: Marine Version of the NH 90 Helicopter (NFH)

Fig. 3.4: German-French Anti-Tank Helicopter PAH-2/HAP/HAC3G

SIMULATEURS D'ETUDES POUR HELICOPTERES

par

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Le CENTRE d'ESSAIS en VOL a pour mission principale d'effectuer les essais en vol officiels d'aéronefs, d'équipement et de systèmes d'armes pour le compte du Ministère de la Défense et le Ministère des Transports.

La répartition géographique du CENTRE d'ESSAIS en VOL est :

- la base principale à BRETIGNY (au sud de PARIS)
- deux bases d'essais à ISTRES et CAZAUX
- deux détachements à TOULOUSE et BORDEAUX.

Depuis le début de ses activités en 1956, la Base d'Essais d'ISTRES est spécialisée dans les essais "avion" et les essais "moteur". Depuis septembre 1986, la section Hélicoptères est également basée à ISTRES.

La longueur de la piste (4000 m) associée à la clémence du climat font de ISTRES un site privilégié pour les essais en vol.

A ISTRES sont également implantés l'Ecole du Personnel Navigant d'Essais et de Réception (EPNER) et le Centre de Simulation.

1 - INTRODUCTION -

L'accroissement de la complexité des systèmes installés à bord des aéronefs nécessite leur étude le plus en amont possible dans le déroulement des programmes. Les gains de temps et d'argent de ces études et des essais menés bien avant les essais en vol sur prototypes ne sont plus à démontrer.

Afin de réaliser ces études, des moyens sols importants sont mis en place. Ces outils permettent d'aider à la définition, au développement et à la mise au point des systemes. Un de ces outils, le simulateur piloté, a l'avantage évident de mettre l'opérateur humain dans la boucle. Il est en effet essentiel de pouvoir disposer le plus tôt possible de remarques concernant le système étudié de la part de l'opérateur humain pour lequel il est conçu.

Depuis longtemps, les études avions utilisent de tels simulateurs de recherche et l'Etat vient de mettre en place au Centre de Simulation du Centre d'Essais en Vol à ISTRES un tel simulateur pour toutes les études hélicoptères.

2 - DESCRIPTION DU SIMULATEUR -

2.1. Modèles mathématiques.

2.1.1. Modèle de mécanique du vol.

Les simulateurs d'études nécessitent l'utilisation de modèles dits de connaissance, c'est-à-dire de modèles fondés sur la théorie de la mécanique du vol

Le simulateur d'ISTRES utilise un modèle général de mécanique Hélicoptère dont l'adaptation à un appareil donné se fait par introduction :

- de coefficients aérodynamiques issus d'essais en soufflerie,
- de caractéristiques géométriques et inertielles.

Le modèle général a été développé par l'AEROSPATIALE Division Hélicoptère dont sont issus actuellement trois modèles d'appareils :

- SA 330 PUMA
- SA 365 DAUPHIN
- HAP / HAC / PAH2

Ces modèles couvrent l'ensemble du domaine de vol hautes et basses vitesses, fortes incidences et forts dérapages.

Afin de valider les algorithmes du modèle général, le Centre d'Essais en Vol a réalisé une campagne d'essais sur un PUMA SA 330. La comparaison des résultats d'essais en vol avec la réponse du modèle a permis de le valider.

Les contraintes de temps de calcul lié au temps réel imposent un temps de cycle inférieur à 20 ms afin de ne pas induire de retard perceptible par l'être humain dans la boucle de pilotage.

Modélisation rotors

Le rotor principal et le rotor arrière ont la même modélisation issue de la théorie de l'élément de pale. Dans le cas du Fenestron, on utilise la formulation basée sur celle du rotor classique avec une contraction du flux dépendant de la vitesse.

Modélisation cellule

Les forces et moments aérodynamiques de la cellule sont calculés à partir de coefficients issus d'essais en soufflerie. Ces coefficients sont obtenus à partir d'un balayage séparé en incidence et en dérapage et sans interactions dûes aux rotors.

Modélisation des intéractions

L'intéraction du rotor principal sur l'empennage horizontal est calculée à partir de l'inclinaison du flux et de la position de l'empennage dans le disque de sillage.

L'intéraction du rotor principal sur le fuselage est la création d'une déportance au voisinage du stationnaire.

L'intéraction du rotor arrière sur la dérive est une poussée parasite en stationnaire et une diminution de l'efficacité de la dérive en vol de translation.

2.1.2. Modélisation des moteurs.

La modélisation des moteurs est spécifique de chaque appareil. Le calcul des performances turbine est issu des courbes de puissance réduite et consommation réduite, en fonction du régime et des conditions extérieures de pression et de température. La réponse dynamique est modélisée par une fonction de transfert qui tient compte du temps de réponse en fonction des évolutions.

2.1.3. Modélisation des commandes de vol.

L'ensemble des paramètres lié aux commandes de vol (débattements, efforts, frottements secs et visqueux) est modélisé pour chaque appareil. Seule la modélisation du pilote automatique du SA 330 PUMA est pour le moment couplée au modèle de mécanique du vol.

La modélisation du pilote automatique du HAP/HAC/PAH2 sera réalisée dès qu'une première définition sera retenue.

2.2. Cabines hélicoptères.

Les essais hélicoptères s'effectuent à partir de deux cabines :

- une cabine monoplace SDVEH (Simulateur de Vol et d'Etudes pour Hélicoptères),
- une cabine biplace côte à côte SUPER-FRELON.

2.2.1. Cabine SDVEH.

La cabine SDVEH est une représentation proche du poste pilote de l'hélicoptère HAP quant à sa forme, au volume disponible et aux masques.

Commandes de vol

Le manche cyclique est relié à deux dispositifs électro-hydrauliques (un pour le contrôle longitudinal et l'autre pour le contrôle latéral) afin de restituer les efforts. Ces systèmes de restitution d'efforts sont réglables par logiciel (efforts, hystérésis, frottements secs, frottements visqueux) et permettent de simuler les commandes cycliques de tout type d'hélicoptère.

Le manche collectif et le palonnier sont pour le moment reliés à des vérins électriques non réglables.

Les manches cyclique et collectif sont adaptés pour recevoir les poignées des différentes versions de l'hélicoptère HAP/HAC/PAH2.

Instrumentation

En dehors de l'instrumentation classique secours, la cabine SDVEH est équipée avec tous les moyens de visualisation mis à la disposition du pilote de l'HAP.

- viseur tête haute : le viseur tête haute VH 130 développé par la société THOMSON sera prochainement installé dans la cabine.
- visualisation tête basse : avant de pouvoir disposer des tubes multimodes prévus pour l'HAP/HAC/PAH2, la cabine est équipée d'un écran FCD 55 de la société THOMSON de dimension 5" x 5" multichrome à balayage cavalier.

- <u>viseur/ visuel de casque</u> : la cabine est équipée d'un viseur/visuel de casque développé par la société THOMSON.

<u>Postes de commandes</u>

Un certain nombre de postes de commandes banalisés sont installés en planche de bord et en banquettes afin de pouvoir dialoguer avec les différents systèmes. Ces postes seront remplacés par les faces avant des postes réels dès qu'une première définition sera retenue, afin de permettre en plus de l'étude système de mener à bien l'étude ergonomique.

<u>Environnement cabine</u>

Le siège équipé d'un système électro-mécanique restitue l'environnement vibratoire de l'hélicoptère à partir d'un logiciel de commande dont les paramètres influents sont le régime rotor, le facteur de charge et la position du manche collectif.

Un module de sonorisation restitue les bruits générés par les ensembles mécaniques, les bruits aérodynamiques et les bruits moteur. Ce module sera à terme complété par l'adjonction d'un générateur des bruits armements (canon, missile, roquette) et des alarmes sonores.

2.2.2. <u>Cabine SUPER-FRELON</u>.

La cabine SUPER FRELON provient du prototype du SA 321 SUPER-FRELON qui a été en 1974 transformé en simulateur pour des études générales. A partir de fin 1986, la place droite doit être reconfigurée pour simuler le poste tireur de l'hélicoptère HAP.

<u>Poignée armement</u>

Seules les commandes armement seront mises à la disposition du tireur ; les commandes de vol et les poignées de pilotage ne seront pas simulées. Les poignées spécifiques armement permettent la gestion temps réel du système d'arme et sont celles dans un premier temps définies pour l'hélicoptère de servitude de l'HAP (HELIOS).

<u>Instrumentation</u>

Il n'y a aucune instrumentation classique. La cabine est équipée des écrans de visualisation mis à la disposition du tireur de l'HAP.

- <u>Visualisations tête basse</u> : 2 écrans tête basse sont installés. L'un est identique à celui de la cabine SDVEH (FCD 55), l'autre est multimode (balayage cavalier et vidéo) et de dimension identique (VSM 55 MH). Ces écrans seront modifiés ou remplacés en fonction des matériels retenus pour l'HAP.
- <u>viseur/visuel de casque</u> : le matériel identique à celui de la cabine SDVEH peut être installé dans la cabine SUPER-FRELON.

<u>Postes de commandes</u>

Comme pour la cabine SDVEH, avant qu'une première définition HAP soit retenue, seuls des postes de commandes banalisés sont installés.

<u>Environnement cabine</u>

Un module de sonorisation identique à celui de la cabine pilote est installé dans la cabine tireur.

2.3. Pupître de direction des essais.

Une salle entièrement instrumentée permet de diriger les essais mais également d'intéragir dans les essais. Elle est phoniquement reliée aux deux cabines qui sont elle-mêmes reliées entre elles.

Cette salle est dotée de moyens de contrôle permettant d'avoir la recopie de tous les écrans des cabines SDVEH et SUPER-FRELON.

L'ingénieur d'essais peut suivre l'évolution de n'importe quel paramètre hélicoptère ou système en numérique ou en graphique.

Un dernier moyen de visualisation permet de suivre l'évolution en 3 dimensions de plusieurs appareils dans le cadre de scénarios de combat air-air, et de visualiser l'environnement vu par l'un quelconque des pilotes.

L'ingénieur d'essais dispose de commandes qui lui permettent de gérer la simulation (mise en attente, arrêt, modifications des conditions de vol,...) et des commandes spécifiques qui lui permettent d'intervenir au niveau de l'essai. Il peut également jouer le rôle de l'un ou l'autre des membres de l'équipage puisqu'il dispose des commandes de pilotage et de tir.

2.4. Environnement.

2.4.1. Visualisation du monde extérieur.

visualisation à maquette

L'image du monde extérieur projetée au pilote ou/et au tireur provient d'une maquette de terrain via une caméra de télévision qui filme à travers une tête optique.

La maquette de terrain est représentative d'une région de FRANCE. Elle mesure 12 m de long et 4 m de large et est à l'échelle 1/1000 ème. Le pilote peut évoluer sur cette maquette dans des conditions identiques à celles du vol tactique.

La tête optique permet de restituer les trois degrés de liberté en rotation (tangage, roulis, lacet) et les trois autres degrés de liberté en déplacement sont restitués par le chariot mobile sur lequel est montée la caméra.

Les performances de débattement angulaire ne sont pas limitées en roulis ni en lacet et sont limitées en tangage à + 18° et - 54°.

Un prisme rajouté à l'ensemble permet de réduire la visibilité jusqu'au vol sans visibilité.

La tête optique est équipée d'un détecteur infra-rouge de proximité pour le plan horizontal et d'un palpeur mécanique pour le plan vertical.

Génération synthétique d'images

Afin de restituer correctement le monde extérieur vu de jour ou de nuit par le pilote ou par le tireur en vision directe ou par l'intermédiaire de tous les capteurs optroniques, il est nécessaire d'utiliser une génération synthétique d'images.

Les caractéristiques de la génération synthétique d'images développée par la Société THOMSON et dont disposera le centre de simulation sont :

- 3 canaux d'imagerie,
- bandes spectrales multiples (vision directe, bas niveau de lumière, infra-rouge et proche infra-rouge,
- points de vue multiples (pilote, tireur, capteurs),
- restitution sur écran multimode à grande résolution (air-sol et appontage).

Cette génération sera opérationnelle fin 1987.

2.4.2. Restitution du monde extérieur en cabine SUPER-FRELON.

L'image du monde extérieur est présentée au tireur dans la cabine SUPER-FRELON sur un moniteur télévision à travers un bloc optique. Ce bloc optique est composé d'une lame semi-réflechissante et d'un miroir sphérique afin de collimater l'image à l'infini.

Le tireur peut également voir le monde extérieur avec le tube tête basse multimode.

2.4.3. Restitution du monde extérieur en cabine SDVEH.

La cabine SDVEH est installée à l'intérieur d'un écran sphérique de 10 m sur lequel il est possible de projeter 3 images différentes se complétant.

A l'intérieur de la sphère les 3 projecteurs suivants sont montés sur une structure parfaitement positionnée :

- projecteur d'horizon,
- projecteur de sol,
- projecteur de cible

Le projecteur d'horizon envoie deux images, une de sol et une de ciel, issues de deux diapositives, à travers deux optiques "fish eyes" de 180° de champ chacune. L'intersection de ces deux images crée l'horizon qui est asservi aux attitudes de l'hélicoptère. L'image projetée ne fournit aucune information de position ou de déplacement.

Le projecteur de sol projette dans une position frontale et fixe une image de sol suivant un format télévision avec un champ de 60° de diagonale.

Le projecteur de cible permet d'envoyer suivant les configurations une image de cible ou de sol dans une quelconque direction à l'intérieur de la sphère.

Configuration Air-Air

L'image du projecteur de cible provient d'une cible analogique (cf paragraphe 2.5.4). Le projecteur dispose d'un zoom qui associé au zoom de la caméra de prise de vue (cf § 2.5.4) permet d'obtenir un grossissement global de 60. Deux optiques interchangeables permettent de couvrir les plages de distance 20 m - 1200 m ou 100 m - 6000 m.

Le projecteur de sol est utilisé seulement pour les phases de combat en basse altitude.

Le projecteur d'horizon est toujours utilisé.

Configuration Air-Sol

Le projecteur de sol n'est pas utilisé.

Le projecteur de cible est utilisé pour envoyer l'image du sol. L'orientation de ce projecteur est commandée par la ligne de visée du viseur de casque. Cette utilisation permet au pilote de voir le sol dans la direction de son regard.

Le projecteur d'horizon est toujours utilisé.

2.4.4. Cible.

L'image cible provient de deux maquettes. L'une est accrochée par l'avant et l'autre par l'arrière, ceci afin d'éviter les masques dûs au système d'accrochage. Ces deux cibles, animées par calculateur, ont trois degrés de liberté en rotation. Elles sont filmées par une caméra de télévision munie d'un zoom et dotée d'un prisme mobile en rotation donnant un 4ème degré de liberté supplémentaire qui permet ainsi d'améliorer les performances du système en vitesse de roulis. Le choix entre l'une ou l'autre cible est géré par logiciel et se fait par allumage et/ou extinction des chambres noires où se trouvent les cibles.

Ces cibles peuvent être animées par différents modes de pilotage :

Scénarios : un certain nombre de scénarios schématiques ont été programmés.

Pilotage simplifié : la cible est animée suivant des trajectoires simples sélectionnées à partir du pupître de direction des essais. Les trajectoires suivantes sont possibles :

- . vol rectiligne en palier,
- . montée ou descente à taux constant,
- . virages à taux constant.

Pilotage au minimanche : le minimanche installé au pupître de direction des essais permet de piloter la cible.

Logiciel de combat : un logiciel de commande automatique permettant à la cible d'évoluer en fonction du contexte chasseur-cible est en cours de développement.

2.4.5. Mouvement.

La cabine SDVEH peut être installée, pour certains essais (qualités de vol...), sur un mouvement à 6 degrés de liberté.

Les performances de ce mouvement développé par la société THOMSON sont :

- . charge utile : 10 tonnes
- . débattements

vertical	:	$\pm$ 1,37 m
longitudinal	:	$\pm$ 1,85 m
latéral	:	$\pm$ 1,60 m
en roulis	:	$\pm$ 28°
en tangage	:	- 31° + 34°
en lacet	:	$\pm$ 29°

.. vitesses	verticale	± 1 m/s
	angulaires	± 25°/s
. accélérations	verticale	± 2,5 g
	longitudinale	± 1,5 g
	latérale	± 1,5 g
	en roulis	± 400°/s²
	en tangage	± 500°/s²
	en lacet	± 400°/s²

2.5. Calculateurs.

2.5.1. Calculateur de simulation.

Le calculateur de simulation est un GOULD SEL 32/7780 bi-processeur de puissance 1,5 Mips. Tous les logiciels de simulation (mécanique du vol, systèmes, gestion des moyens d'environnement...) sont implantés dans ce calculateur qui doit être remplacé par un calculateur plus performant (7 Mips).

En outre, le Centre de Simulation dispose d'un calculateur identique réservé au développement des programmes.

2.5.2. Calculateur générateur de symboles.

Les symboles présentés sur les visualisations pilote tireur et ingénieur d'essais (viseur tête haute, têtes basses, viseur de casque) sont issus de deux ensembles SINTRA CONCEPT 60 de la Société CIT ALCATEL, l'un affecté aux visualisations pilote, l'autre aux visualisations tireur, les visualisations du pupître de direction des essais étant alimentés en recopie par les deux ensembles. Ces deux systèmes sont couplés au calculateur GOULD de simulation.

2.5.3. Boîtier Générateur de Symboles Pilotable (BGSP).

La société THOMSON développe pour le centre de simulation de nouveaux générateurs de symbôles. Ces calculateurs sont capables de générer de la symbologie pour tout type de visualisation (balayage cavalier ou vidéo) avec un logiciel très voisin de celui des boîtiers générateurs de symbôles embarquables, avec les mêmes performances et sans aucun problème de couplage puisque réalisés à partir des mêmes matériels.

Cet outil permettra de valider les logiciels de symbologie avant les phases d'essais en vol et surtout permettra tout transfert des essais vol vers les essais sol et vice-versa instantanément pour étude ou modification.

3 - OBJECTIFS D'ETUDES -

Le simulateur du Centre d'Essais en Vol à ISTRES permet les études :

- aide à la conception du véhicule,
- aide à la conception des systèmes,
- organisation des postes d'équipage,
- évaluation opérationnelle.

Pour les années à venir, le potentiel du simulateur sera presque entièrement consacré aux études liées au programme HAP/HAC/PAH2. Ce paragraphe se limitera donc à traiter des études liées à ce programme.

3.1. Conception du véhicule.

A partir du modèle de mécanique dont les coefficients sont issus des essais en soufflerie et des résultats de calcul, le simulateur piloté permet de fournir rapidement des remarques de la part des pilotes et de pouvoir influer sur la conception du véhicule.

En particulier les essais de qualités de vol sont essentiels pour connaître le comportement de l'hélicoptère et évaluer son aptitude en tant que porteur à la bonne exécution de la mission et en particulier au vol tactique.

Cette aide à la conception peut influer sur :

- le rotor principal,
- le rotor arrière,
- l'empennage,
- les commandes de vol.

L'étude des pannes
. rotor arrière
. empennage (si braquable)
. hydraulique,

peut également conduire à des modifications dès la conception.

3.2. Conception des systèmes.

Système stabilisateur et pilote automatique

Le simulateur permet d'étudier et d'évaluer les différents modes du pilote automatique adaptant la maniabilité de l'hélicoptère aux besoins du pilote et ceci principalement en vol tactique. Ces études permettent de mettre au point les lois de pilotage et les lois d'efforts aux commandes.

Il faut également étudier un mode du pilote automatique couplé au système d'arme pour contrer les effets dûs au tir canon.

Régulation moteur

Des essais dans des conditions de vol tactique permettent d'étudier le comportement de la régulation moteur (stabilité et temps de réponse) et d'analyser son influence sur le pilotage hélicoptère.

Système d'arme

Ces études doivent permettre d'évaluer les performances des algorithmes de conduite de tir. Ces performances concernent l'efficacité du tir dans les différentes conditions de vol de la cible et du chasseur ainsi que l'aspect pilotabilité.

Les moyens dont dispose actuellement le Centre de Simulation permettent d'évaluer la conduite de tir canon, missile ou roquette, mises en oeuvre par le pilote ou le tireur avec le viseur de casque ou le viseur tête haute. Les essais liés au viseur principal nécessitent l'utilisation de la génération synthétique d'images.

3.1. Organisation des postes d'équipage.

Sous ce terme général se regroupent les études

- d'interface équipage - système,
- de dialogue pilote - tireur

qui peuvent se décomposer suivant les thèmes

- ergonomie,
- symbologie,
- modes et commandes.

Ergonomie

Les études ergonomiques d'un cockpit sont essentielles et doivent être à la base de toute étude d'interface équipage - système.

Ceci nécessite un maquettage réaliste des postes pilote et tireur présentant une grande souplesse d'adaptation et de modification.

L'ensemble des études suivantes est à mener :

- commandes de vol, lois d'efforts, déplacements, seuil....
- poignées : préhension, position des commandes, choix des commandes (bouton poussoir, sélecteur, palette, instable...), efforts des commandes, accessibilité...
- organisation des planches de bord : répartition des instruments, lisibilité, accessibilité,
- postes de commandes (viseurs, armement, gestion systèmes...), accessibilité, lisibilité, facilité de mise en oeuvre,
- viseurs, implantation, lisibilité, facilité de mise en oeuvre, interférence entre les viseurs :
  . pilote : viseur clair - viseur de casque,
  . tireur : viseur principal - viseur de casque.
- tubes tête basse : lisibilité, accessibilité des commandes liées à ces visualisations.

Toutes ces études doivent être menées dans un premier temps point par point puis ensuite dans un contexte global du vol. En effet, les aspects ergonomiques du contrôle hélicoptère ainsi que ceux du contrôle du tir nécessitent simultanément la presque totalité de tous ces points. Ces études doivent être faites dans des conditions de vol tactique de jour et de nuit en tenant compte des problèmes d'éclairement, d'éblouissement, de reflet et de visibilité avec les jumelles à bas niveau de lumière.

Les alarmes qui font partie d'une étude plus générale qui est celle des pannes ont un aspect ergonomique fondamental. Il est en effet essentiel d'avertir l'équipage sans délai et sans ambiguïté de la panne qui vient de survenir et de l'action à entreprendre. L'étude ergonomique des alarmes consiste en la détermination du type d'alarme (visuelle, sonore, vocale , sensorielle) à activer pour alerter et informer au mieux l'équipage.

Symbologie

L'étude des symbologies qui a également un aspect ergonomique consiste en la définition des informations symboliques présentées au pilote et au tireur sur chacun des supports de visualisation dont ils disposent :

. viseur tête haute,
. viseur de casque,

. viseur principal,
. écrans tête basse,
. postes de commandes et visualisations,

en respectant la cohérence et l'homogénéité entre les écrans et en sélectionnant les informations nécessaires et suffisantes pour chaque phase de vol.

- pilotage . vol tactique de jour et de nuit
  . vol en conditions IMC
  . vol de nuit avec image thermique.
- navigation . vol tactique,
  . vol en conditions IMC.
- observations, visée et tir . viseur de casque
  . viseur tête haute
  . viseur principal.

La symbologie qui est la réponse du système dans l'interface équipage - système est également le support essentiel dans le dialogue pilote - tireur.

Modes et commandes

Les modes et commandes sont les entrées de l'interface équipage - système d'arme. Cette étude est absolument essentielle en simulation car elle a des répercussions importantes sur l'architecture du système.

Pour l'hélicoptère HAP ce système est très complexe puisqu'il y a 2 membres d'équipage disposant chacun de 2 viseurs et mettant en oeuvre 3 types d'armenent (canon, missile, roquette).

Le découpage artificiel dû à l'exposé écrit ne correspond pas à un besoin lié aux études et aux évaluations. Au contraire, les études d'ergonomie, de symbologie et de modes et commandes sont menées simultanément, chacun des thèmes ayant des répercussions sur les autres.

Les essais sont effectués par tranches correspondant à des niveaux de définition, chaque tranche se terminant avec les phases de mise au point par une évaluation opérationnelle.

3.4. Evaluation opérationnelle

C'est la phase finale des essais (ou des tranches d'essais) où, à partir de scénarios opérationnels correspondant à des missions on évalue la réponse de l'ensemble véhicule - système et équipage face aux besoins opérationnels.

Ainsi, pour chaque scénario, il faut évaluer :

. ergonomie des commandes,
. dialogue équipage - système,
. dialogue pilote - tireur
. influence du pilotage sur la visée,
. influence du tir sur le pilotage,
. performances de tir,
. quantification de la charge de travail.

4 - CONCLUSION -

Ce document présente les moyens de simulation mis en place au Centre d'Essais en Vol à ISTRES au profit des études hélicoptère. L'importance des moyens ainsi que leurs futures extensions à court terme font de ce simulateur un outil essentiel pour le développement des hélicoptères futurs.

Opérationnel depuis 1984, ce simulateur a déjà permis de mener des études au profit de l'hélicoptère HAP/HAC/PAH2. Un vaste programme d'essais mis en place pour les années à venir permettra de recueillir de précieuses données avant les essais en vol des hélicoptères de servitude puis des prototypes.

MODELING XV-15 TILT-ROTOR AIRCRAFT DYNAMICS BY FREQUENCY AND TIME-DOMAIN IDENTIFICATION TECHNIQUES

by

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SUMMARY

Models of the open-loop hover dynamics of the XV-15 Tilt-Rotor Aircraft are extracted from flight data using two approaches: frequency-domain and time-domain identification. Both approaches are reviewed and the identification results are presented and compared in detail. The extracted models compare favorably, with the differences associated mostly with the inherent weighting of each technique. Step responses are used to show that the predictive capability of the models from both techniques is excellent. Based on the results of this study, the relative strengths and weaknesses of the frequency and time-domain techniques are summarized, and a proposal for a coordinated parameter identification approach is presented.

NOMENCLATURE

p,q,r	roll rate, pitch rate, and yaw rate, respectively, deg/sec
u,v,w	longitudinal, lateral, and vertical velocities, respectively, m/sec
γ^2_{xy}	coherence function between variable x and y
$\delta_a,\delta_e,\delta_r$	aileron surface deflection (deg), elevator surface deflection (deg), and rudder surface deflection (deg), respectively
δ_c	power lever deflection, %
ζ	damping ratio
τ	time delay
ϕ,θ,ψ	roll, pitch, and yaw angles, respectively, rad (deg)
ω	undamped natural frequency, rad/sec
$1/T$	inverse time constant, rad/sec

1. INTRODUCTION

Dynamics identification methodologies generally fall into two categories: frequency-domain and time-domain. The choice of techniques to be used is usually based on the analyst's personal familiarity with the methods and on the specific application. Each approach has inherent strengths and weaknesses. Frequency-domain identification uses spectral methods to determine frequency responses between selected input and output pairs. Then, least-squares fitting techniques are used in the frequency-domain to obtain closed-form analytical transfer-function models of linear input-to-output processes. Time-domain identification first requires the selection of a state-space model structure, which may be linear or nonlinear. Model parameters are identified by least-squares fitting of the response time-histories or by maximum likelihood methods. Transfer functions for linear models and frequency responses can then be obtained from the identified state-space formulation.

The US Army has been developing frequency-domain identification techniques in support of handling qualities, flight, and simulation experiments. Extensive flight experiments have been conducted on the

XV-15 Tilt-Rotor Aircraft (Fig. 1). References 1 and 2 present the identified open-loop frequency responses, transfer functions, and model verification results. Frequency-domain identification tests have also been recently conducted on the Bell-214-ST single (teetering) rotor, and CH-47 tandem-rotor aircraft (Refs. 3 and 4). In the Federal Republic of Germany, the DFVLR has had extensive experience with maximum-likelihood, time-domain identification techniques. Linear and nonlinear model-identification methods have been developed. Much of the DFVLR experience with helicopter identification has been associated with the highly coupled BO-105 hingeless-rotor helicopter (Refs. 5 and 6).

As part of an ongoing US/FRG Memorandum of Understanding (MOU) on helicopter flight control, an extensive joint study is being conducted to analyze the XV-15 data-base for the (open-loop) hover flight condition using both time- and frequency-domain techniques. The primary objectives of this study are to: (1) gain a better appreciation for the relative strengths and weaknesses of each technique; and (2) develop improved methods of identification for rotorcraft.

This paper reviews the dynamics identification techniques which have been developed in the US and the Federal Republic of Germany. The results of applying these techniques to the XV-15 data base are presented and compared, and sources of differences in the extracted models are discussed. Finally, conclusions concerning the appropriate applications for each technique and proposals for unified identification methods using both approaches are presented.

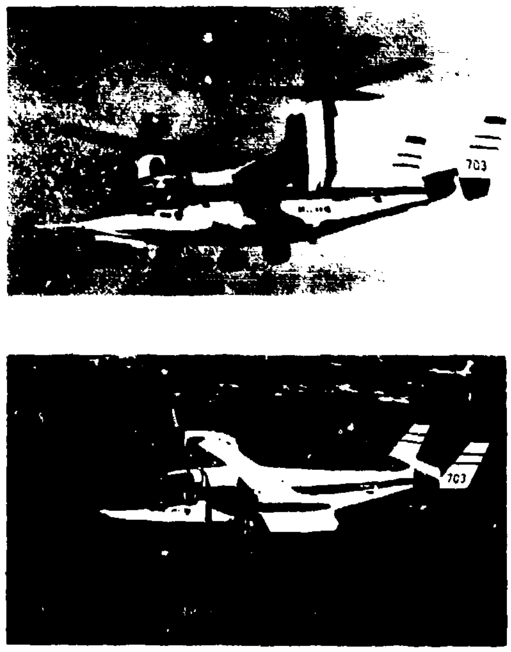

Fig. 1. The XV-15 Tilt-Rotor Aircraft. a) Hover configuration; b) cruise configuration.

2. OVERVIEW OF FREQUENCY-DOMAIN AND TIME-DOMAIN IDENTIFICATION TECHNIQUES

A. Frequency-Domain Identification Method

The frequency-domain identification approach developed by the US Army is depicted in Fig. 2. Spectral methods based on the Chirp z-transform are used to extract high-resolution frequency responses between selected input and output pairs. The identification results are presented in Bode-plot format: magnitude and phase of the output to the input versus frequency. These identification results are nonparametric because no model structure has been assumed. As such, they can be very useful for flight-control system design and handling-qualities compliance testing; for example, currently proposed handling-qualities criteria for the LHX (Ref. 7) are based on frequency-domain parameters which can be read directly from these graphical results. Frequency responses obtained from real-time and nonreal-time simulations can be compared directly with the flight data to expose limitations and discrepancies in the simulator models (Ref. 1). The fact that this comparison can be made initially without an a priori assumption of model structure or order is especially important for verifying mathematical models of new aircraft configurations. When the model structure and parameter values are required, they may be obtained by fitting the tabulated frequency-responses with analytical transfer-function models to extract modal characteristics. Examples of this application are the testing of handling-quality specifications given in

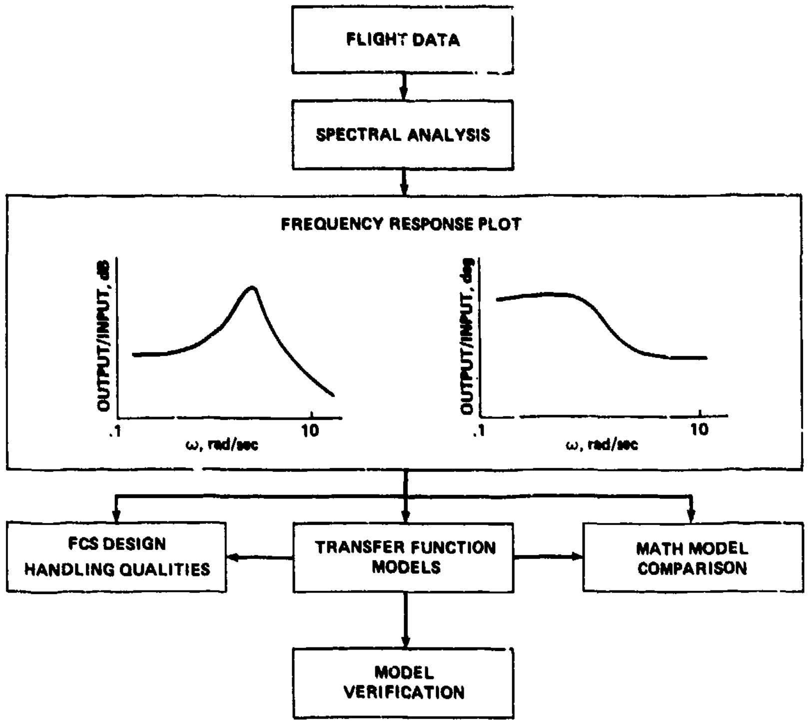

Fig. 2. Frequency-domain identification method.

lower-order equivalent system terms, and the examination of transfer function-based control system designs. Since this fitting procedure is completed after the frequency response is extracted, the order of the transfer function can be carefully selected to avoid an overparameterized model. Multi-input/multi-output frequency-response methods are suitable for extracting a transfer matrix which includes the important coupling effects. Finally, the extracted models are driven with the flight-test control inputs to verify the time-domain response characteristics.

The semilog frequency format of the Bode-plot presentation and subsequent transfer-function fit makes the identified transfer-function and state-space models most accurate at mid and high frequency (initial time history transients). The low-frequency and steady-state response prediction of the extracted models is generally not as good as in time-domain identification approaches.

B. Time-Domain Identification Method

The general approach used in time-domain identification is shown in Fig. 3. Time-based identification techniques are initially applied to the data to check their internal compatibility. Data inconsistencies resulting from calibration errors, drifts, or instrumentation failures are detected by comparing redundant measurements from independent sensors, such as rate and attitude gyros, or altitude change and vertical acceleration (Ref. 6). This approach, which can be used on-line, helps to ensure that only consistent data are generated for the further evaluation and system identification.

For this next step, the aircraft dynamics are modeled by a set of differential equations describing the external forces and moments in terms of accelerations and state and control variables. The coefficients in these equations are the stability-derivatives. In some cases, a priori values for these derivatives can be obtained from analytical calculations, wind-tunnel data, or from start-up identification techniques such as a least-squares method. The responses of the model and aircraft resulting from the flight-test control inputs are then compared. The response differences are minimized by the identification algorithm which iteratively adjusts the model parameters. In this sense, aircraft system identification implies the extraction of physically defined aerodynamic and flight mechanics parameters from flight-test data. Usually, it is an off-line procedure since some skill and iteration are needed to develop an appropriate model formulation. Model formulation involves consideration of model structure, selection of significant parameters, and inclusion of important nonlinearities. Time-domain techniques yield a multi-input/multi-output model that is appropriate for application in stability and control analysis, simulation, and control system design. The identified parameters are also useful for comparison and correction of analytically or wind-tunnel derived stability-derivatives (Ref. 8).

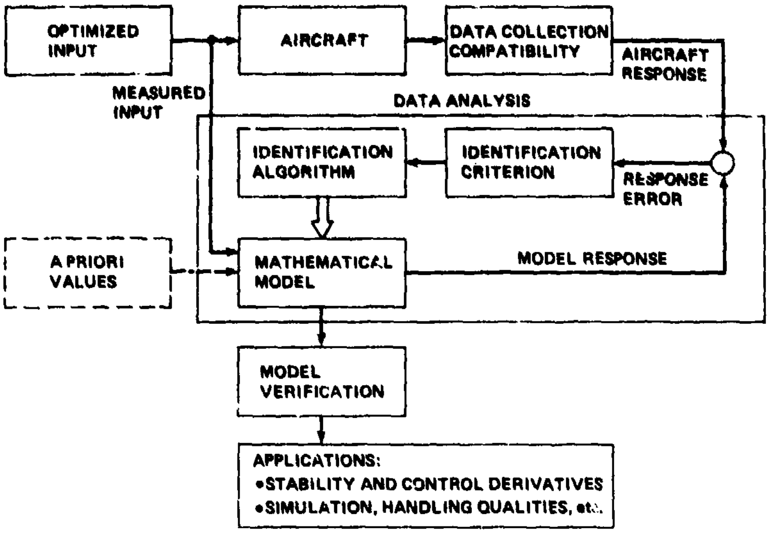

Fig. 3. Time-domain identification method.

A key feature of the time-domain identification technique is that the extracted models are based on the curve fitting of the original measured (time-domain) flight-test data. Errors which may occur in the transformation of the data from the time- to the frequency-domain are thus avoided. The identified models can then be easily presented in the frequency-domain as Bode plots or, if the identified model is linear, also as parametric transfer-functions.

3. IDENTIFICATION OF XV-15 OPEN-LOOP DYNAMICS IN HOVERING FLIGHT

This section reviews the XV-15 flight-test data base for parameter identification and presents and compares the results of (linear) frequency and time-domain identification methods for the open-loop hover flight condition. For illustrative purposes, the roll response identification is discussed in detail.

A. Flight-Test Data Base

The complete data base for dynamics identification includes four flight conditions from hover to cruise. The present study concentrated exclusively on the identification of the open-loop hover dynamics because:

The dynamics for this flight condition are coupled and very unstable which makes this case the most difficult to analyze.

Nonlinear effects are the most significant in the hover flight condition, which allows a good demonstration of the nonlinear identification techniques developed by the DFVLR.

Focusing on the rotor-borne flight condition maximizes the carry-over of the present experience to future rotorcraft identification studies to be carried out under the MOU.

The pitch and roll axis instabilities for the hover flight condition are characterized by a time-to-double amplitude of about 3 sec. Therefore, long-period inputs needed to identify the low-frequency vehicle dynamics are not practical for the open-loop hovering vehicle. Extraction of the open-loop vehicle dynamics from closed-loop testing is possible subject to an important condition: the total surface deflection, which is comprised of inputs from the pilot and the stability and control augmentation system (SCAS), must contain a significant component which is uncorrelated with the response of the vehicle (Ref. 9). Then, the required low-frequency inputs can be conducted on the closed-loop (stable) vehicle.

Flight-Test Inputs. Two types of inputs were executed in the identification flight tests. "Frequency-sweep" inputs were used for model extraction, and step inputs were used for model verification.

Two typical concatenated lateral stick frequency-sweeps completed during the hover flight tests of the XV-15 are shown in Fig. 4a. These tests used pilot-generated rather than computer-generated inputs. The sweep is initiated with two low-frequency input cycles corresponding to the lower bound of the frequency range of primary interest (0.2-6.0 rad/sec). These cycles ensure good excitation of the low-frequency vehicle dynamics. After the initial two low-frequency cycles, the lateral stick is oscillated at progressively higher frequencies for an additional 50 sec. By the end of the 90 sec duration test, the stick is being driven at fairly high frequencies (4 Hz shown in Fig. 4). The input amplitudes are fairly

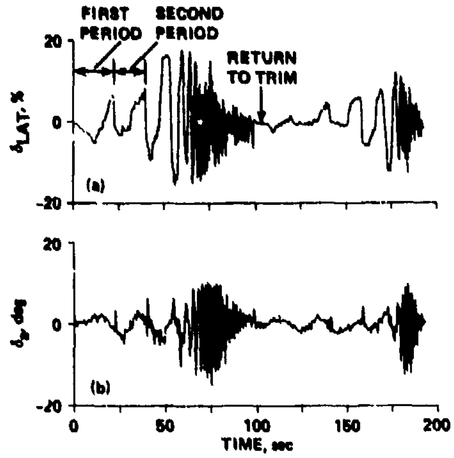

Fig. 4. Two lateral stick frequency-sweeps (δ_{LAT}) in hover. a) Lateral stick inputs; b) aileron surface deflections.

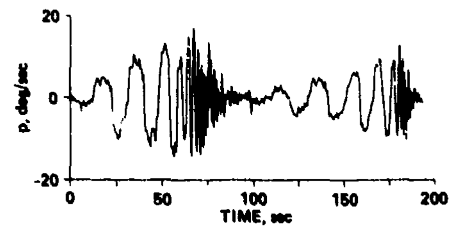

Fig. 5. Roll-rate response (p) during lateral frequency-sweeps.

small at low frequency where vehicle motions are considerable, with larger inputs at mid frequency, and smaller inputs again at very high frequency. The associated aileron surface deflection (total input to the aircraft‡) shown in Fig. 4b reflects a significant component from the pilot input (note that δ_a and δ_{LAT} are defined with opposing sign conventions). The resulting roll-rate amplitudes of 10-20 deg/sec as shown in Fig. 5 are typical for frequency-sweep tests. The frequency-sweep is especially well suited for frequency-domain identification because it is a periodic input form that excites the vehicle in all of its dominant modes of motion within the frequency range of interest. This input also has some advantages for time-domain identification. Vehicle excitation is restricted to be within the frequency-range of model applicability which is especially important for meaningful state-space parameter results (Ref. 4). Also, the monotonic increase in frequency allows the time-domain identification to be frequency-weighted which compensates for the inherent low-frequency weighting of this method.

Step inputs are commonly used in the flight test community to expose dominant vehicle characteristics, so they represent a good test of the identified model's predictive capability. Step inputs were executed in both the open- and closed-loop condition. Open-loop verification ensures that the identified models reflect the dynamics of the open-loop vehicle and not those of the inverse feedback element (Ref. 9). Step inputs with the flight-control system engaged are also useful since the steadier initial conditions allow fine differences between the model and the flight responses to be exposed.

B. Frequency-Domain Identification

The most important step in the frequency-domain identification procedure is the extraction of accurate, high-resolution frequency responses between the various input and output pairs. A key metric for assessing the quality of the frequency-response identification is the coherence function (γ^2_{xy}). This frequency-dependent function indicates that fraction of the output response which is linearly related to the excitation signal. The random error associated with the frequency-response identification is dependent on the value of the coherence function at each frequency, and on the number of (independent) time history segments ("windows," N_d):

‡Although the aileron, elevator, and rudder surfaces are not effective in hover, they continue to be actuated in addition to the primary effectors which are the rotor collective and swashplate controls. It was found to be most expedient to refer all the transfer functions to these surface deflections, since neglecting the small servo lags, these are related to the sum of the pilot and SCAS inputs through a mixing ratio which is constant across the entire flight envelope.

$$\varepsilon = \frac{[1 - \gamma_{xy}^2]^{1/2}}{|\gamma_{xy}|\sqrt{2n_d}} \quad (*) \tag{1}$$

The length of the window (secs) determines the amount of low frequency power and the associated low-frequency coherence which can be achieved. Low variance in the spectral identification therefore requires high coherence and multiple concatenated time-history records. Initial analyses of the XV-15 data base used all available repeat runs (three were used in the original analysis of Ref. 1), without concern of the individual coherence quality of each run. Subsequent time-domain analyses by the DFVLR and frequency-domain analyses by the US Army indicated that some of the frequency-sweep runs were unsuitable for identification, and should be removed from the concatenation procedure. In the case of the lateral axis frequency-sweep, one of the three runs was found to be unsuitable because of low coherence. The frequency-response obtained with the remaining two (good) runs has substantially improved spectral quality. This frequency response and the associated coherence function are shown in Figs. 6 and 7. Good identification is achieved over the frequency range 0.2-9.0 rad/sec.

The magnitude response peak is due to the dominant roll modes which are in the frequency-range of 0.5-1 rad/sec; the associated phase rise indicates that the modes are unstable. At the higher frequencies (1.0-10 rad/sec), the magnitude and phase plots follow a K/s characteristic. The value of the constant (K) is the roll response sensitivity (L_{δ_a}). The relatively flat phase response at high frequency indicates a very small value of effective time delay. Finally, the drop in coherence function near the magnitude peak suggests the existence of nonlinearities for large vehicle motions.

(1) Lateral/Directional Transfer-Function Models

The selection of the order and structure of the transfer-function models is predominantly based on three important factors (Ref. 4):

(a) The models must be appropriate to the frequency-range of concern (0.2-6.0 rad/sec in the present study).

(b) The models must provide a reasonable fit of the input-to-output frequency response within the frequency range associated with good coherence.

(c) The selected models should be based on a theoretical analysis of the effective physical order of the system. Therefore, the appropriate transfer-function models are a function of flight condition and flight-control system status (i.e., SCAS-on or SCAS-off).

For the open-loop XV-15 in the hovering flight condition, the yaw (and heave) responses are essentially decoupled and first order in nature. Therefore, an appropriate model for yaw-rate response to pedal inputs is:

$$\frac{r}{\delta_r}(s) = \frac{N_{\delta_r} e^{-\tau_\psi s}}{(s + 1/T_y)} \tag{2}$$

The on-axis roll (and pitch) responses are dominated by the hovering cubic, and as seen in Fig. 6 have one excess pole at high frequency:

$$\frac{p}{\delta_a}(s) = \frac{L_{\delta_a} s(1/T_{\phi_1})(1/T_{\phi_2}) e^{-\tau_\phi s} \quad (+)}{(1/T_y)(1/T_r)[\zeta_r, \omega_r]} \tag{3}$$

The dominant source of coupling in the open-loop configuration is the yaw response to lateral stick inputs. This coupling arises from the rotor torque differential which accompanies the differential collective inputs used for roll control. Frequency-response identification of the coupled response (Ref. 1) indicates an appropriate transfer-function model of:

$$\frac{r}{\delta_a}(s) = \frac{N_{\delta_a}(1/T_\psi)[\zeta_\psi, \omega_\psi] e^{-\tau_{\psi_2} s}}{(1/T_y)(1/T_r)[\zeta_r, \omega_r]} \tag{4}$$

The denominator parameters of the lateral/directional transfer-function models (Eqns. 2-4) represent natural dynamics modes of the vehicle. Therefore, the common modes must have the same values for all three responses. Maintaining this relationship is essential for achieving unique and physically meaningful transfer-function models. While it is possible to fit all three responses simultaneously to maintain the commonality of denominator parameters, this approach is not the best. A better strategy is to identify individual parameters from the on-axis frequency-response in which they have the dominant effect.

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*Window overlapping further reduces the random error below that shown in Eqn. (1) (Ref. 9).
+Shorthand notation: $[\zeta, \omega]$ implies $s^2 + 2\zeta\omega s + \omega^2$, ζ = damping ratio, ω = undamped natural frequency (rad/sec); and $(1/T)$ implies $s + (1/T)$, rad/sec.

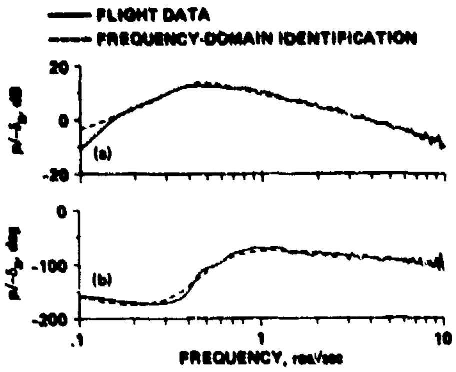

Fig. 6. Frequency-domain identification of roll-rate response to aileron ($p/-\delta_a$). a) Magnitude; b) phase.

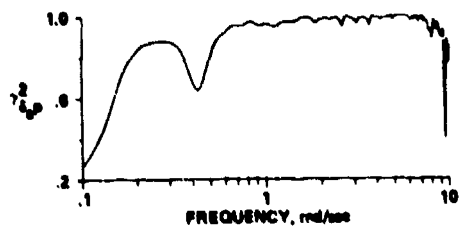

Fig. 7. Coherence function for roll-rate response identification ($\gamma^2_{\delta_a p}$).

Then these parameters are fixed in the identification of the off-axis transfer-functions. So, for example, the following yaw response transfer-function is obtained from the pedal sweeps:

$$\frac{r}{\delta_r}(s) = \frac{0.619\ e^{-0.0210s}}{(s + 0.102)} \tag{5}$$

This result shows that the yaw response of the tilt-rotor configuration is very lightly damped, as compared to a standard helicopter with a tail rotor. The small effective time delay indicates that lags caused by unmodeled high-frequency dynamics are negligible.

The next step is to identify the roll-rate transfer function (p/δ_a). Since the yaw mode has been identified in Eqn. 5, this parameter is fixed in the roll-response transfer function (Eqn. 3). Then the remaining parameters are varied to obtain the best least-squares fit of the roll rate frequency-response (Fig. 6):

$$\frac{p}{\delta_a}(s) = \frac{-3.71s(-0.107)(0.412)e^{-0.0313s}}{(0.102)(1.23)[-0.418,\ 0.447]} \tag{6}$$

The fitting range is from 0.2-9.0 rad/sec, in which the coherence function ($\gamma^2_{\delta_a p}$) indicates good spectral accuracy. As expected from the phase response characteristics (Fig. 6b), the open-loop roll-response dynamics are dominated by an unstable roll mode with the frequency of about 0.4 rad/sec. The associated time-to-double amplitude is 3.5 sec. The pole-zero pair $(1/T_{\phi 1})/(1/T_y)$ is at very low frequency and nearly cancels out. This reveals that yaw coupling does not noticeably affect the roll-response characteristics. Therefore, a lower-order roll response model which contains only the hovering (lateral) cubic roots $(1/T_r)[\zeta_r,\omega_r]$ and entirely ignores yaw coupling is an appropriate approximation for this vehicle. This assumption is common for hovering aircraft. The low-frequency numerator factor $(1/T_{\phi 1})$ associated with lateral translation damping, is marginally unstable (time-to-double amplitude = 7.5 sec) indicating a very low value of the velocity damping derivative (Y_v). Finally, the effective time delay for the roll response (τ_ϕ) is small, suggesting that, as in the yaw response (Eqn. 5), the unmodeled high-frequency lags are not significant.

With the lateral/directional denominator factors (dominant vehicle modes) identified using the on-axis frequency-responses, the numerator factors of the off-axis response (r/δ_a) can now be extracted. The denominator factors of Eqn. 4 are fixed and the least-squares fit gives:

$$\frac{r}{\delta_a}(s) = \frac{0.344(-0.345)[0.868,\ 0.487]e^{-0.00900s}}{(0.102)(1.23)[-0.418,\ 0.447]} \tag{7}$$

In the frequency range $\omega > 1.2$ rad/sec, the yaw response to aileron inputs is dominated entirely by the coupling derivative, N_{δ_a}. At low frequencies, the dynamics are affected by the unstable hovering cubic.

As shown in Figs. 6a and 6b, the transfer-function model of Eqn. 6 is a good representation of the identified roll response in the range of satisfactory coherence (0.2-9.0 rad/sec). Although the present transfer-function model (Eqn. 6) is not significantly different from that obtained previously (Ref. 1) using all of the available sweep runs (including the poor quality runs), the match between the model and flight data is significantly improved.

Close examination of Fig. 6 shows that the match between the model and flight data is much better in magnitude than in phase. This is because of the relative weighting selected for the identification (1 DB magnitude error: 7° phase error) which is common for lower-order equivalent system matching (Ref. 10). On the basis of the steep phase response of the flight data at the dominant mode (ω = 0.5 rad/sec) as compared to the transfer-function model, a lower (less negative) damping ratio is indicated. This inconsistency between the magnitude and phase responses indicates nonlinear behavior in the dominant modes of roll motion. As mentioned previously, this is also reflected by the drop in coherence in the same frequency range. Significant side-by-side nonlinear rotor interactions are known to exist for large lateral-velocity transients, as were encountered during the low-frequency inputs.

The lateral/directional transfer-function model results of this section are summarized in Table 1.

(c) Longitudinal Transfer-function Models

In hovering vehicles, pitch and roll dynamics are analogous. The pitch response is dominated by a longitudinal hovering cubic analogous to the lateral hovering cubic, and a first-order heave response is analogous to the first-order yaw response. Power lever (vertical control) and longitudinal stick inputs do not induce significant inter-axis coupling in the XV-15 configuration.

Spectral analysis of the individual pitch-sweeps showed that only one of the three runs had satisfactory coherence for use in identification. (The original analysis of Ref. 1 used all three runs.) Transfer-function models are extracted from the identified frequency responses using the same approach discussed above for the lateral/directional dynamics. The heave response is determined first from collective sweeps, and then the pitch response is determined with the identified heave mode ($1/T_h$) fixed. The resulting transfer functions for pitch rate and vertical acceleration responses are summarized in Table 1. As in the roll case, the pitch response is dominated by a hovering (longitudinal) cubic, comprised of a low-frequency unstable oscillation, $[\zeta_p, \omega_p]$, and a stable aperiodic mode ($1/T_p$). Also, the pitch transfer-function model fits the identified frequency response much better in magnitude than in phase. Based on phase response considerations alone, the unstable damping ratio would, as before, be much lower (less negative). The discrepancy between the magnitude and phase fits is again due to nonlinearities associated with the large velocity perturbations encountered during the low-frequency inputs.

C. Time-Domain Identification

Maximum likelihood (ML) technique is generally accepted as one of the most suitable time-based methods for aircraft parameter identification. The main advantages of the ML estimation are:

(1) It yields asymptotically unbiased and consistent estimates for linear systems.

TABLE 1 Comparison of Transfer-Function Models for Hover

Frequency-Domain Identification	Time-Domain Identification
$\frac{r}{\delta_r}(s) = \frac{0.619\, e^{-0.0210s}}{(0.102)}$	$\frac{r}{\delta_r}(s) = \frac{0.732\, e^{-0.0320s}}{(0.0987)}$
$\frac{p}{\delta_a}(s) = \frac{-3.71s(-0.107)(0.412)e^{-0.0313s}}{(0.102)(1.23)[-0.418,\ 0.447]}$	$\frac{p}{\delta_a}(s) = \frac{-3.53s(0.072)(0.106)e^{-0.0320s}}{(0.0987)(0.830)[-0.242,\ 0.461]}$
$\frac{r}{\delta_a}(s) = \frac{0.344(-0.345)[0.868,\ 0.487]e^{-0.00900s}}{(0.102)(1.23)[-0.418,\ 0.447]}$	$\frac{r}{\delta_a}(s) = \frac{0.353(0.658)[-0.0540,\ 0.240]e^{-0.0320s}}{(0.0987)(0.830)[-0.242,\ 0.461]}$
$\frac{a_z}{\delta_c}(s) = \frac{-0.00980s\, e^{-0.00740s}}{(0.105)}$	$\frac{a_z}{\delta_c}(s) = \frac{-0.00953s\, e^{-0.0320s}}{(0.122)}$
$\frac{q}{\delta_e}(s) = \frac{-2.66s(-0.271)(0.508)e^{-0.0656s}}{(0.105)(1.32)[-0.463,\ 0.579]}$	$\frac{q}{\delta_e}(s) = \frac{-2.30s(0.0280)(0.119)e^{-0.0320s}}{(0.122)(0.808)[-0.272,\ 0.499]}$

units:
p, q, r : deg/sec
a_z : g
$\delta_a, \delta_e, \delta_r$: deg
δ_c : %

(2) It provides the Cramer-Rao-Bound, which is a measure of the reliability of each estimate.

(3) It yields the correlation between the identified parameters.

Both the Cramer-Rao-Bound and the parameter correlation help to develop an appropriate model structure and to avoid "over-parameterization." A nonlinear maximum likelihood method developed by DFVLR (Refs. 11, 12) was utilized for time-domain identification of the XV-15. This technique allows a general linear and nonlinear model formulation of the state and measurement equations:

$$\dot{x}(t) = f(x(t), u(t) - \Delta u, s) \qquad x(t = 0) = x_0$$
$$y(t) = g(x(t), u(t) - \Delta u, s) + \Delta y \tag{8}$$

where

x = computed state vector
y = measured variables
u = measured control vector
x_0 = initial conditions
s = system parameters
$\Delta u, \Delta y$ = zero shifts

The initial conditions and zero shift terms are included to compensate for drifts and offsets in the measurements.

For the XV-15 data evaluation, the ML program was first utilized to check compatibility of the measurements, and to reconstruct the nonmeasured data. Then the program was used for the parameter identification itself. In the following sections, these steps are discussed in detail.

a. Data Compatibility and Reconstruction

The XV-15 instrumentation system provides attitude rates, attitude angles, and linear accelerations; speed measurements for the hover flight condition are not available. Therefore, only the compatibility of the angular data could be evaluated. A satisfactory agreement between calculated and measured angles was found and no additional corrections were made. For the frequency sweeps, speed components were derived by integrating the measured linear accelerations. Since, for these tests, the aircraft is in trim at the beginning and the end of each sweep, speed equation biases can be estimated to meet the boundary conditions: $u(0) = v(0) = w(0) = u(t_F) = v(t_F) = w(t_F) = 0$. For the system identification, the calculated velocity variables are included in the measurement vector together with the linear accelerations. Strictly speaking, these derived data do not provide additional information about the system dynamics; however, they help to keep the speed response of the model within a realistic range and to prevent long-term speed drifts. This characteristic is important since the identification procedures requires the integration of highly unstable (hover) system differential equations for a time duration of about 90 sec.

b. Identification of the Lateral/Directional Motion

Preliminary time-domain identification analyses showed that the longitudinal and lateral/directional motions of the XV-15 are practically decoupled. The main emphasis was placed on the identification of a linear lateral/directional model. This model is represented by (linear) differential equations for the lateral force, rolling moment, and yawing moments. The general 3 DOF model is:

$$\dot{x} = Ax + Bu + bx$$
$$y = Cx + Du + by \tag{9}$$

where

$x^T = (v, p, r, \phi)$
$y^T = (a_y, v, p, r, \phi)$
$u^T = (\delta_a, \delta_r)$

The unknown coefficients in the state matrices (A and C) and the control matrices (B and D) are the desired stability and control derivatives. The bias vectors bx and by are estimated constants representing drifts and zero shifts.

In each flight test, a controller (either δ_a or δ_r) was used to excite the aircraft modes. To obtain sufficient information about both roll and yaw motion for the identification, data obtained from an aileron and a rudder sweep were combined. This "multiple run evaluation" yields one common model for both runs (except for the bias terms, which must be estimated for each individual run). This approach has been used successfully in previous helicopter identification studies (Ref. 5).

Three main characteristics of the XV-15 lateral/directional dynamics became obvious from the initial identification analysis:

The yaw motion which is due to rudder inputs is virtually decoupled and the significant parameters, yaw damping and the control derivative, can easily be extracted from the rudder-sweep data. Yaw-model and aircraft time histories are in good agreement.

There is some coupling from the aileron inputs to the yaw motion. Therefore, the control-coupling derivative N_{δ_a} was included for identification.

For the aileron-sweep data, it was not possible to obtain a satisfactory curve fit for the total run duration. The major difficulty is the identification of the roll-moment equation and, consequently, the fit of the roll-rate response.

The third characteristic caused some severe identification problems and was investigated in more detail.

One approach to this problem was to use shorter time intervals of the aileron sweep (only the low- or mid- or high-frequency part). With this approach, the responses of the identified models fit the measured roll rates almost perfectly. However, there were major differences in the estimated parameters from the original identification based on the total run duration. Tests with different a priori values to start the ML technique were also made to ensure that the ML criterion did not lead to local minima (a common identification problem). Results from these calculations clearly showed that the data contain strong nonlinearities which cannot be described by a linearized model. One logical next step is the extension of the model to include the appropriate nonlinearities; this extension will be addressed later. Another possibility is to stay with a linear model, accept its deficiencies, and define its range of validity and applicability. This approach is discussed first.

Lateral/directional model identification was conducted separately on the three available aileron-sweep runs, each in combination with a rudder-sweep run. When the total run duration was used, all three sweep results showed the same tendency:

The model response matched the low-frequency part of the data fairly well.

The model response underestimated the flight data as the input frequency increased, with up to a 50% error in roll rate at high-input frequencies.

These results make sense in light of the ML identification criterion:

$$L = \sum_{i=1}^{N} [(z(t_i) - y(t_i))^T \cdot R^{-1} \cdot (z(t_i) - y(t_i))] + N/2 \cdot \ln |R| \tag{10}$$

where

N = number of data points
z = measurement vector
y = model response vector
R = measurement noise covariance matrix

The optimum is reached when the differences between the amplitudes of the measured and calculated time histories are minimized. From Fig. 4b, it is seen that about 70% of the total run duration of the aileron-sweep is low-frequency data. Consequently, the identification method emphasizes primarily the longer-duration, low-frequency part of the data, and sacrifices the accuracy of the shorter, high-frequency part. For many applications, the initial and short term (higher-frequency) response of a system is of more interest than the long term (lower-frequency) behavior. Therefore, it was desirable to improve the identification result for the higher-frequency range, allowing larger errors for the low frequencies. Methods to meet this objective are:

i. Conduct frequency-sweeps with more emphasis on the high-frequency content.

ii. Apply alternate control inputs (e.g., multisteps) which excite mostly the mid- and high-frequency dynamics.

iii. Use only the higher-frequency sweep data for the identification.

These approaches were either not possible (new flight testing required for options i and ii) or they were felt to be a poor compromise (iii).

Another solution is to increase the influence of the amplitude errors for a selected part of the data. When frequency-sweep inputs are used, this can be done by the "multiple segment evaluation": a part of the data (e.g., high-frequency range) is treated as a separate test. It is combined several times with the original test data so that, in principal, the weighting of the chosen data points is arbitrarily increased. This approach worked satisfactorily, but it yielded an increased number of unknown biases, needed more data handling and, in particular, required more computing time. But pursuing this basic idea, the identification program was modified to allow different weighting of selected time periods within one run. This approach turned out to be very efficient as it does not require estimating any additional parameters, or computing capacity.

The data weighting technique was applied for the identification of the aileron frequency-sweeps. Increased weights were used for the roll-rate fit errors which occurred in the higher-frequency part of the data. Good agreement of the measured and calculated data was thus obtained for the mid- and high-frequency inputs, whereas there were larger discrepancies for the low-frequency inputs. Also, there was good consistency of results for the three repeat runs. The results also confirmed that it is advantageous to keep the low-frequency data in the evaluation as they provide the necessary speed-derivative information. The mean values and standard deviations for the derivatives obtained from the identification of the aileron sweeps are summarized in Table 2. As time-domain techniques tend to be sensitive to phase shifts, a time lag for the control input was estimated as a multiple of the sampling rate. In state-space format, the final time-domain identification (mean-value) model for the lateral/directional motion is:

$$\begin{bmatrix} \dot{v} \\ \dot{p} \\ \dot{\phi} \\ \dot{r} \end{bmatrix} = \begin{bmatrix} -0.0749 & 0 & 9.81 & 0 \\ -0.0179 & -0.559 & 0 & -0.349 \\ 0 & 1 & 0 & 0 \\ 0.00140 & 0 & 0 & -0.0715 \end{bmatrix} \begin{bmatrix} v \\ p \\ \phi \\ r \end{bmatrix} + \begin{bmatrix} -0.0112 & 0 \\ -0.0617 & 0 \\ 0 & 0 \\ 0.00615 & 0.0128 \end{bmatrix} \begin{bmatrix} \delta_a \\ \delta_r \end{bmatrix}^* \tag{11}$$

units: v : m/sec
p, r : rad/sec
ϕ : rad
δ_a, δ_r : deg

*Time delay in control input is $\tau = 0.0320$ sec.

Figure 8 gives the time-history comparison for one of the sweeps with the state-space model. Once again, this final time-domain model correlates well at medium and high frequency, with some discrepancy at low frequency. Overall, however, the agreement is quite satisfactory.

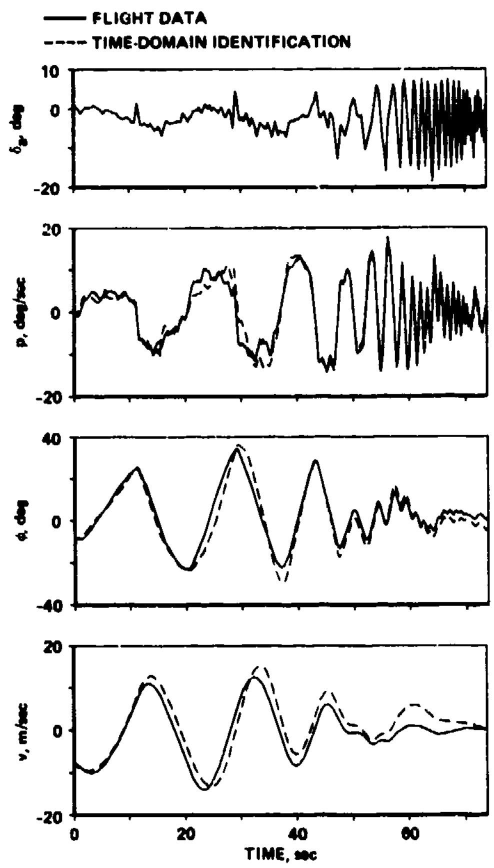

Fig. 8. Time-domain identification of lateral/directional model.

The present XV-15 study is the first experience with explicit data weighting. This technique certainly requires further development and, as with all such weighting methods, should be used very carefully. In this regard, the comparison with the frequency-domain results is very helpful in evaluating the confidence and the range of validity of the results.

c. Identification of the Longitudinal Motion

For the identification of the longitudinal model, an elevator sweep test was combined with a power sweep test. The identification results for the longitudinal dynamics were analogous to the preceding lateral/directional results:

The heave equation is practically decoupled and can easily be identified from the power sweep tests to obtain the vertical damping and the control derivatives.

The elevator sweep showed the same tendency as the aileron sweeps: it was not possible to determine a model that is equally good for the low- and high-frequency range. Again, the main problem occurred in the moment equation so that the discrepancies were seen in the pitch-rate comparison.

Only one of the three available flight tests could be evaluated. When the other two tests were used, the identification results diverged and became unusable. This is in agreement with the frequency-domain analysis which also indicated some problems with these runs. For the one remaining elevator sweep run, the first 30 sec of data had to be removed in order to reach convergence in the estimation. Again, the data-weighting technique was successfully used to obtain a satisfactory fit for the higher frequency part of the data. The longitudinal model parameters are given in Table 2. Unfortunately, the identified longitudinal model is based on only a rather limited amount of data. Therefore, except for the heave equation, this model cannot be expected to have the same level of reliability as the lateral/directional model. However, the good comparison with the frequency-domain results as discussed in the next sections show that the time-domain model accurately represents the XV-15 longitudinal dynamics.

TABLE 2 [illegible]e-Domain Identification Results

Derivative	Mean	Variance	Standard Deviation	Standard Deviation (% of mean value)
a) Lateral/Directional Paramet[illegible] runs)				
Y_v	-0.0749	2.04 × [illegible]	0.0143	-19.1
L_v	-0.0179	5.67×10^{-6}	0.00238	-13.3
Y_{δ_a}	-0.0116	1.41×10^{-7}	0.000376	-3.37
L_p	-0.559	1.03×10^{-2}	0.101	-18.1
N_v	0.00141	3.45×10^{-6}	0.00186	132.
L_r	-0.349	1.33×10^{-2}	0.115	-33.0
L_{δ_a}	-0.0617	3.00×10^{-6}	0.00173	-2.81
N_{δ_a}	0.00615	2.15×10^{-7}	0.000463	7.53
N_r	-0.0715	6.00×10^{-6}	0.00245	-3.42
N_{δ_r}	0.0127	1.78×10^{-7}	0.000422	3.30
b) Longitudinal Parameters (1 run)				
X_u	-0.0636			
X_w	0.0175			
X_{δ_e}	0.0939			
Z_u	-0.0685			
Z_w	-0.122			
Z_{δ_e}	-0.0469			
Z_{δ_c}	-0.00959			
M_u	0.0204			
M_w	-0.00160			
M_q	-0.477			
M_{δ_e}	-0.0401			

units:
- u, v, w : m/sec
- p, q, r : rad/sec
- $\delta_a, \delta_e, \delta_r$: deg
- δ_c : %
- X, Y, Z : n
- L, M, N : n-m

4. COMPARISON OF IDENTIFICATION RESULTS

This section compares the frequency- and time-domain identification results. This comparison is done in frequency- and time-domain formats, since both are important for ensuring model fidelity. In the frequency-domain format, transfer-function parameters from the frequency-domain identification are compared with those obtained from the state-space formulation. Also, frequency responses from the two models are compared with the flight-data frequency response. Since the frequency-domain format is the "natural environment" for frequency-domain identification, the models obtained from this approach generally fit the flight-data responses better than those obtained from time-domain identification. Comparison in the time-domain format is achieved by driving the frequency-domain models with the frequency-sweep input histories. The resulting responses are compared with the responses of the vehicle and the time-domain identification fits. Since this is the "natural environment" for time-domain identification, models identified using this approach generally match the flight data better here. (The detailed discussion of the results for the roll-axis is continued, and the results for the remaining axes are again summarized.)

A. Comparison in the Frequency-Domain Format

Transfer functions are obtained from the time-domain identification results of Eqn. 11 (and Table 2) by Cramer's rule, and are tabulated for comparison with the frequency-domain results in Table 1.

(1) Lateral/directional Models

The results of Table 1 indicate that the lateral/directional modes (denominator factors of the transfer functions) are nearly identical for both techniques, except for the difference in the unstable damping ratio (ζ_R). The high-frequency gain and time delay of the three transfer-functions compare very well, while there are some differences in the low-frequency numerator factors.

The relative significance of the differences in the transfer-function parameters can be more clearly seen from the frequency-response comparison of the models with the flight data. The roll responses of the identified models and the aircraft are shown in Fig. 9. At frequencies of $\omega > 1$ rad/sec, both models correspond almost exactly with the flight data. Also, both models correctly predict a low-frequency instability at $\omega = 0.5$ rad/sec, with a falling magnitude response for lower frequencies.

A closer examination of the magnitude and phase comparisons shows that the frequency-domain identification result matches the magnitude-response curve better than the time-domain identification result in the vicinity of the dominant mode ($\omega = 0.5$ rad/sec). However, the time-domain identification result matches the phase-response curve better in this frequency-range. This difference is due entirely to the inherent weighting of the two methods. In the frequency-domain identification method, the relative weighting between magnitude and phase is arbitrary, but the standard choice (1 dB error: 7° error) has produced satisfactory results in a number of identification studies conducted by one of the authors (Refs. 1-4). In time-domain identification, the performance index is much more sensitive to phase errors, which generate a large area between the model and flight-data responses. Thus, the phase response is more closely matched. Also, time-domain identification results can be highly sensitive to the identified value of time-delay, which must be an integral multiple of the sample rate.

The fact that magnitude and phase curves cannot be matched simultaneously in either frequency or time-domain identification methods further indicates the existence of important nonlinearities in the low-frequency roll oscillation modes. Therefore, linear models (from either method) are a compromise and cannot fully characterize the nonlinear behavior of the vehicle. Both methods capture the important vehicle response characteristics and are generally in good agreement with each other. Similar agreement is also exhibited in the yaw responses to rudder (r/δ_r) and aileron (r/δ_a) inputs.

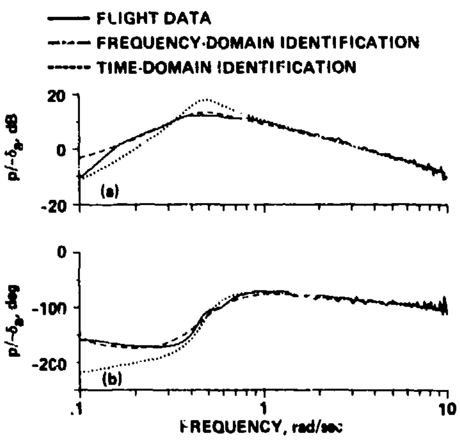

Fig. 9. Comparison of roll-response models ($p/-\delta_a$) in the frequency-domain format. a) Magnitude; b) phase.

(2) Longitudinal Models

The comparison of transfer-function models for the longitudinal degrees-of-freedom is very similar to the preceding results for the lateral/directional degrees-of-freedom. The dominant modes of motion for the two methods are very close, except for the difference in the unstable damping ratio, ζ_p (again roughly a factor of 2). The high-frequency gain and time delays of the two transfer functions are also nearly identical, with some differences in the low-frequency numerator parameters. As before, the frequency-response match between the two models and the flight data is nearly identical for frequencies greater than 1 rad/sec. Also, in the frequency range near the dominant mode (ω = 0.5 rad/sec), the frequency-domain model fits the magnitude response better, while the time-domain model fits the phase response better. Once again, nonlinearities and differences in inherent weighting of the methods is the cause of this discrepancy. In general, however, the agreement between the models and the flight data is quite satisfactory.

B. Comparison in the Time-Domain Format

Transfer functions obtained from the frequency-domain identification were converted into a canonical state-space representation to generate time histories for the comparison with the flight-test data. A bias term was estimated for each equation (using a least-squares procedure) to compensate for zero shifts and drifts. Figure 10 shows an aileron-sweep time history compared with the frequency-domain identified model, and the time-domain identified model. For the high-frequency inputs, both models yield virtually the same result and agree with the flight-test data. In the lower frequency range, some differences between the two models and differences with the flight-test data can be seen. Generally, the agreement with the flight data is quite satisfactory, so it can be stated that both identified models represent the dynamics of the aircraft fairly well. The discrepancies between the two model responses, however, indicate that no unique model can be identified; the slightly different results reflect the specific identification criterion of each method. This confirms the preceding conclusions from the comparison in the frequency-domain format.

5. TIME-DOMAIN VERIFICATION USING STEP-RESPONSE DATA

A good way to judge the utility of the identification results is to compare the prediction of the identified models with the vehicle response for inputs other than those which were used in the identification procedure. Here, step inputs are used since these are quite different from the frequency-sweep forms which are used in identification. (These step inputs tended to be very rounded in nature, so low-pass, preconditioning to remove high-frequency elements of the input signal is not necessary as was done in Ref. 4.)

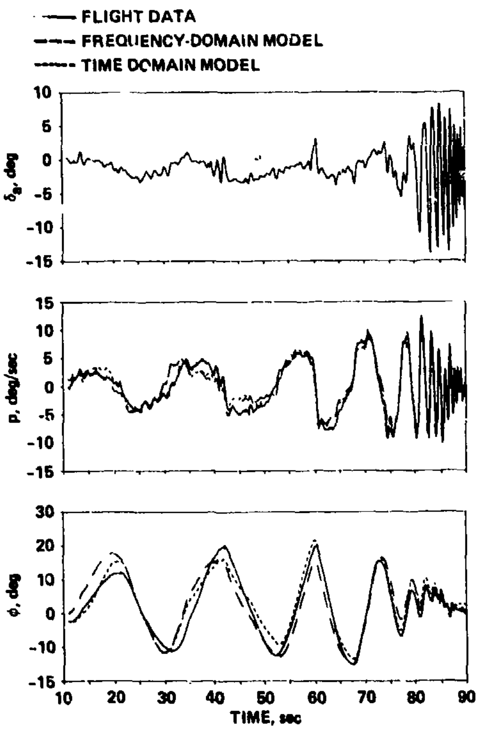

Fig. 10. Comparison of roll-response models in the time-domain format.

A. Lateral/Directional Motion

The identification of the yaw motion did not cause any difficulties. Therefore, a satisfactory prediction capability can be expected. Figure 11 compares the yaw-rate model responses with the measured data for a pedal-step input. Good agreement is apparent for both models. The small discrepancies are caused by an inaccurate calibration factor between pedal and rudder (surface) deflection, and some mid-frequency mismatch of the first-order yaw model (Ref. 9).

Identification problems associated with the aileron-sweep evaluation have been discussed in detail. The verification using step input data offers a good possibility to check the validity of the linear models. Lateral stick step-inputs were flown with the roll SCAS-off and the yaw SCAS-on; so the measured yaw-rate response shown in Fig. 12 results from the pilot's lateral stick and pedal inputs, and the yaw-SCAS activity. The comparison of the roll rate (p) and roll angle (ϕ) response proves that both models are able to predict accurately the aircraft motion. This agreement is also true for the yaw rate comparison which indicates that the coupling derivative (N_{δ_a}) was correctly identified. Minor differences between the two model responses probably result as before from the different weighting methods.

B. Longitudinal Motion

The heave response is practically decoupled and gave no problems in either identification method; good verification results are expected. Figure 13 shows that the power step responses agree with the measured (quite noisy) vertical acceleration data. The responses are shown separately for the two models since they are practically identical and cannot be distinguished when shown within the same plot.

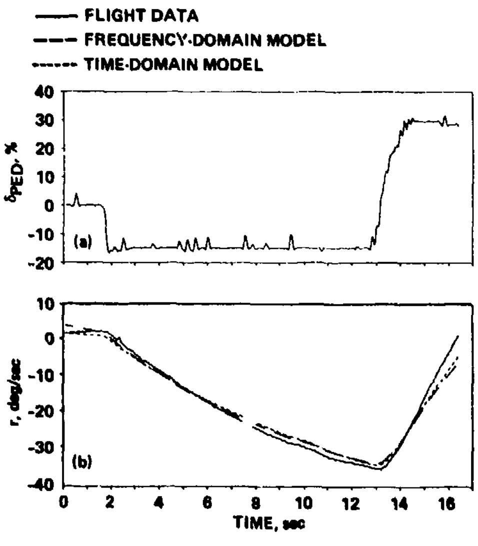

Fig. 11. Comparison of yaw-rate response prediction for step rudder input (yaw SCAS-off). a) Pedal input ±100% δ_{ped} = ±44 deg δ_r; b) yaw-rate response.

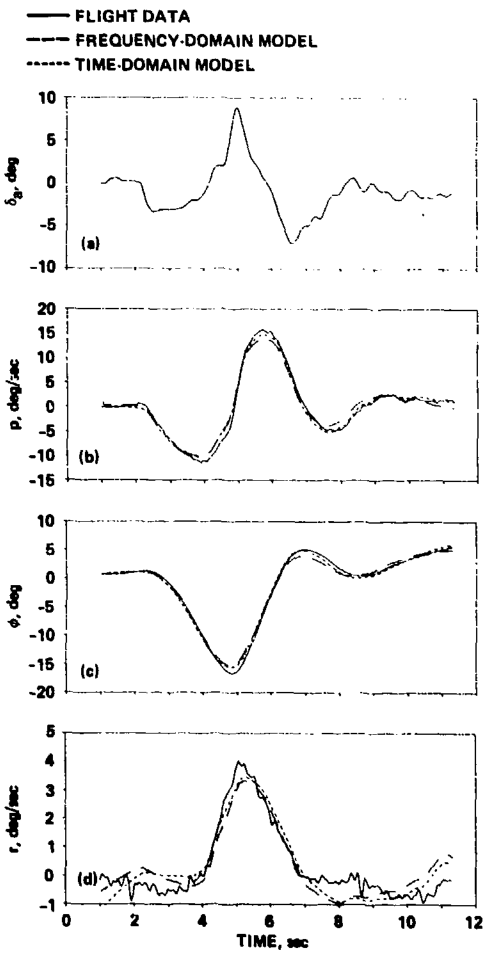

Fig. 12. Comparison of lateral/directional response prediction for step aileron input (roll SCAS-off, yaw SCAS-on). a) Aileron input; b) roll rate; c) roll angle; d) yaw rate.

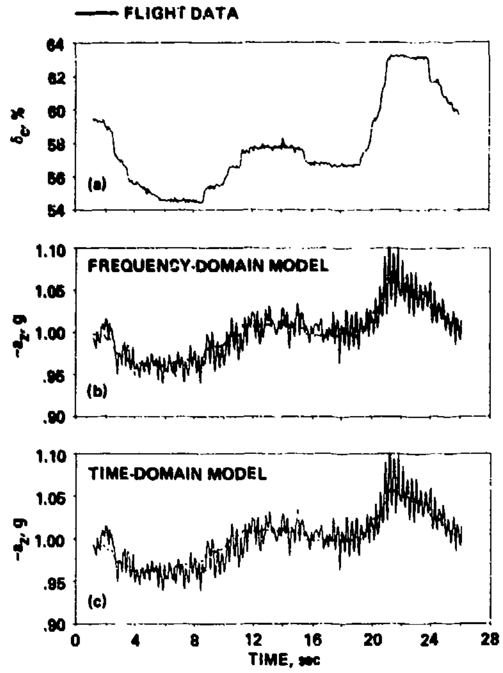

Fig. 13. Comparison of vertical-acceleration response prediction for step power lever input. a) Power lever input; b) frequency-domain model; c) time-domain model.

Nonlinearities and a limited pitch-response data base led to problems in identifying a longitudinal model, as has been discussed. Therefore, the verification tests using elevator steps are particularly helpful in checking the reliability of the two models. Figures 14 and 15 compare the pitch-model responses for two different flight conditions: pitch SCAS-off and pitch SCAS-on. In both cases, the identified models yield a good prediction of the aircraft response. Again, the minor differences between the responses are due to the inherent weighting of each method.

The SCAS-off data fit of Fig. 14 is of special interest. Since the models were extracted from SCAS-on flight-test data, some output/input correlation cannot be avoided and may lead to significant identification errors; in the worst case, the inverse feedback transfer function rather than the open-loop aircraft response would be identified (Ref. 9). However, the good agreement between the model time histories and the SCAS-off flight data in Fig. 14 clearly demonstrates that the <u>open-loop dynamics of the aircraft</u> were determined. The overall excellent correlation of the models and step-response data adds confidence to the accuracy of identified derivatives and transfer functions, and the estimation techniques.

6. NONLINEAR MODEL IDENTIFICATION

The preceding identification results from both time- and frequency-domain techniques have demonstrated that linear model identification yielded a compromise between low- and high-frequency data fits, or between magnitude- and phase-response fits. They suggest the existence of significant nonlinearities, particularly in the roll and pitch axes. Therefore, the nonlinear maximum likelihood time-domain method was utilized to identify an extended model and to investigate the importance of various parametric terms.

Relatively large amplitude aircraft responses during the low-frequency inputs (see, for example, Fig. 8) are a common characteristic of all of the frequency-sweep flight-test data. Deviations from the steady-state trim are in the range of:

9-14 m/sec in lateral speed (aileron-sweep)

8-11 m/sec in longitudinal speed (elevator-sweep)

25-37 deg in roll angle (aileron-sweep)

17 deg in pitch angle (elevator-sweep)

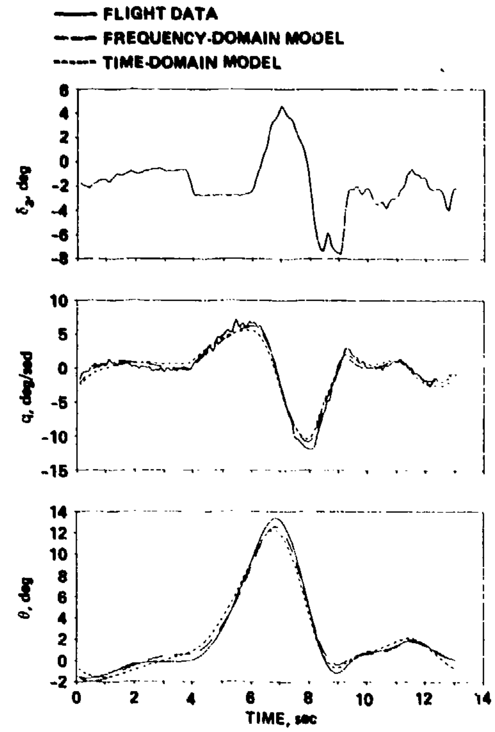

Fig. 14. Comparison of pitch-response prediction for step elevator input (pitch SCAS-off). a) Elevator input; b) pitch rate; c) pitch angle.

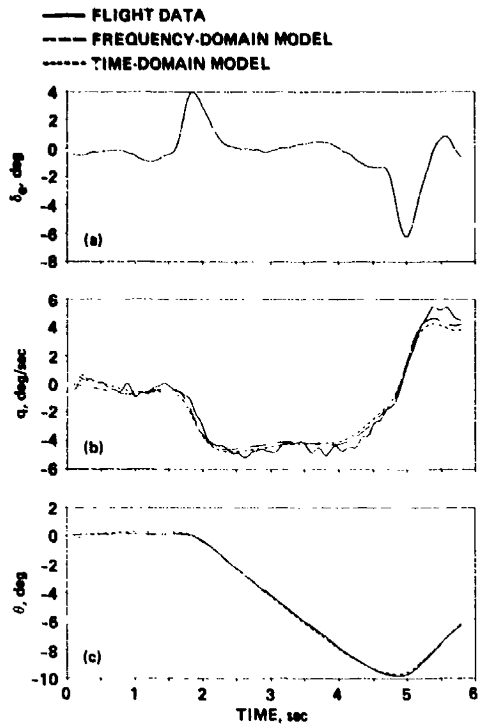

Fig. 15. Comparison of pitch-response prediction for step elevator input (pitch SCAS-on). a) Elevator input; b) pitch rate; c) pitch angle.

These amplitudes are certainly large enough to violate the small-perturbation assumptions for linear models; further, the dynamic characteristics of hovering rotorcraft are especially sensitive to translational speed changes. Therefore, the linear model was first extended by adding nonlinear speed derivatives (L(v**2) and L(v**3)) to the roll moment equation: the curve-fits improved significantly, in particular owing to L(v**2). Further attempts to reduce the remaining discrepancies were made by including additional nonlinear terms. Their significance was checked with time-history comparisons and the evaluation of the parameter covariance matrix, which indicates the reliability of the identified parameter and the correlation with other parameters. As a preliminary result, a model was identified that includes the above mentioned speed derivatives and, in addition, L(δ_a**2) and L(δ_a*u).

Figure 16 shows that the nonlinear model fits the measured data almost perfectly. The results presented in Fig. 16 are preliminary and are intended to illustrate the possible role of nonlinearities in the dynamics. It is important to note that the model was identified without the use of any explicit data weighting. This suggests that the additional weighting (e.g., high frequency versus low frequency) is not required when an appropriate model formulation is applied. However, the evaluation again revealed a well known identification problem: it is always possible to improve the time-history curve fit by arbitrarily adding model parameters. But, a useful model requires the estimated derivatives to have physical significance. It is the responsibility of the analyst to define and select meaningful additional terms. For the side-by-side rotor configuration of the XV-15, the speed-related derivatives (L(v**2) and L(v**3)) are physically justified. Similarly, the control effectiveness may be in fact nonlinear and dependent on forward speed; but these effects should be further investigated.

7. ASSESSMENT OF IDENTIFICATION METHODS

This cooperative study has provided the unique opportunity for specialists using different methods to compare and coordinate analyses of a common rotorcraft data base. This experience has been invaluable for gaining a better appreciation for the advantages and limitations of both techniques, and for formulating ideas for an integrated approach to dynamics identification. The following observations are based on the results of this cooperative effort.

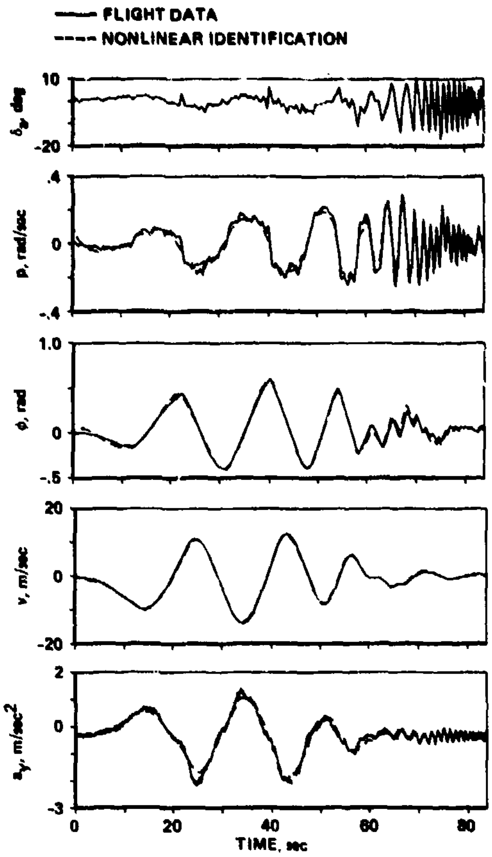

Fig. 16. Example of time-domain identification of a nonlinear lateral/directional model.

The principal advantages of frequency-domain identification are:

a. Frequency responses of the dominant on-axis input/output pairs are rapidly generated and are very useful for gaining a good appreciation for the inherent vehicle dynamics. The fact that the extracted frequency-responses are independent of pre-assumed models is important for the initial assessment of natural system order, dominant-mode locations, and stability characteristics. As a result, a better choice of appropriate model structure and order is possible.

b. Parameters associated with the high-frequency dynamic behavior can be determined directly from the frequency-responses without any a priori assumption of model structure. Specifically, the control derivatives (e.g., $L_{\delta_a}, M_{\delta_e}$) are determined from the high-frequency gain responses, and the equivalent time delays (e.g., τ_ϕ, τ_e) are determined from the high-frequency phase responses.

c. Weighting can be accomplished explicitly. Relative weights can be arbitrarily assigned to the magnitude and phase curves. Model fitting can also be arbitrarily weighted more to the low- or high-frequency range--depending on the intended use of the model.

d. Accurate, high-resolution frequency-response identification is given the main emphasis in this method. CHIRP z-transform methods are especially well suited for identifying frequency-responses from noisy flight data. The resulting transfer-function models are a much closer representation of the frequency-response characteristics than is possible with time-domain identification.

The principal disadvantages of frequency-domain identification are:

a. Current techniques are not well suited for highly coupled multi-input/multi-output (MIMO) system identification, although two-input/single-output identification has been successfully attained in the present study. More highly automated techniques are needed to make the frequency-domain methodology efficient when the required number of input/output frequency-responses is large. Also, methods for simultaneous fitting of many coupled responses is necessary to ensure commonality of transfer-function denominator parameters for MIMO models.

b. Frequency-domain identification results in transfer-function models. Individual stability derivatives are not readily extracted unless the assumed models are of very low order.

c. Spectral analysis assumes input-to-output linearity. For nonlinear systems, the transfer functions are linearized describing functions; identification of pure nonlinear parameters is not possible.

The principal advantages of time-domain identification are:

a. The method is naturally suited to multi-input/multi-output identification since the model can be of arbitrary order and structure. This method is especially well suited for identifying highly coupled systems.

b. Stability and control derivatives are identified explicitly, and the method leads to the identification of a complete state-space model.

c. Considerable effort is invested in achieving the highest quality of time-domain data. Data consistency, drop-out tests and signal-reconstruction methods are an integral part of the time-domain identification procedure. The least-squares fitting in the time-domain with high-quality time-history data results in a much better time-domain fit of the frequency-sweep responses.

d. Extended maximum-likelihood techniques can be used to identify parametric nonlinearities which are especially important in the low-frequency dynamics of hovering rotorcraft.

The principal disadvantages of time-domain identification are:

a. The results are dependent on presumed model structure and order. When a new configuration is being identified, a priori knowledge of model structure may not be available, and considerable variation in the parameters can occur when the model structure is altered.

b. Explicit frequency-domain weighting is not possible. Specifically, the time-domain method inherently weights phase errors more heavily than magnitude errors. This characteristic makes the extracted state-space model very sensitive to pure time delays and unmodeled high-frequency dynamics. Also, the method inherently weights low-frequency dynamics much greater than high-frequency dynamics; this weighting can be adjusted when the input signal has monotonically increasing frequency content as in the frequency-sweep.

c. Confidence in the individual state-space model parameters may be very low since the identified state-space model can contain a high degree of internal cancellation in the overall input-to-output response.

8. A PROPOSAL FOR A COORDINATED FREQUENCY-DOMAIN/TIME-DOMAIN IDENTIFICATION METHOD

The preceding assessment of the advantages and limitations of each identification method suggests the following coordinated frequency-domain/time-domain identification approach:

Step 1. Use time-domain signal conditioning methods to clean up the flight data for drop-outs, wild points, and consistency. For example, rate gyros can be integrated and compared with attitude gyros.

Step 2. Identify the dominant input/output on-axis frequency-response characteristics using only the best runs, as determined from coherence analyses of the individual frequency-sweeps. Identify the effective time delay and high-frequency control sensitivity directly from the frequency-response plots.

Step 3. Formulate low-order system models from inspection of the identified frequency-response plots and theoretical analyses. Determine the on-axis transfer-function parameters.

Step 4. Formulate a state-space model which has a structure and order consistent with the transfer-function model formulation. Time-domain identification should be completed with the equivalent time delay fixed at the value identified in Step 3. Weighting should be applied to the time-history data to ensure that the control sensitivity derivatives are maintained at the value identified in Step 3. (Alternatively, the control derivatives can be fixed.)

Step 5. Compare the extracted on-axis transfer-functions, frequency-responses and time histories from the time-domain and frequency-domain results. If substantial errors exist, go back to Step 2; reevaluate the quality of the spectral-responses, time responses, and the order and structure of the selected models. If the models are found to be the best which can be achieved under the assumption of linearity, pursue nonlinear maximum-likelihood methods to identify the dominant parametric nonlinearities.

Step 6. Verify the extracted models using time-history data from inputs not used in the identification procedure. If significant errors between the predicted and actual response characteristics exist, reevaluate the significance of observed discrepancies in frequency and time-domain identification fits. If necessary, go to Step 3 and increase the order of the models; but, check for the possibility of model over-parameterization by trying a few different verification inputs.

9. CONCLUSIONS

This joint effort has provided the unique opportunity for specialists in different techniques to compare their approaches using a common flight test data base. On the basis of this comparison, it has been shown that the frequency and time-domain methods each have important advantages and inherent limitations. Future identification efforts must be based on a comprehensive, coordinated approach which uses both frequency and time-domain methods.

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OPERATIONAL LOAD MEASUREMENTS ON SERVICE HELICOPTERS

by

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SUMMARY

The philosophy of Operational Load Measurement programmes as applied to rotary wing aircraft is reviewed. A major aim is to substantiate component fatigue lives in the light of operational usage. The aircraft installation to meet the objectives of the programmes is described. The analysis techniques, which are designed to provide the operator and engineer with a visibility of the origin of fatigue damage, are described with illustrative examples taken from current programmes. The Paper draws attention to problem areas that must be addressed in comprehensive fatigue monitoring systems.

1 INTRODUCTION

In the UK, the structural usage of many fixed and rotary wing aircraft in RAF service are assessed through Operational Load Measurement (OLM) programmes. These programmes seek to validate the fatigue substantiation process in the light of operational usage. Fatigue damage due to in-service loading actions is evaluated against a structural performance, which is established through component or major structural fatigue tests, in as direct a manner as possible. In these programmes, a few aircraft of the fleet are comprehensively instrumented with strain gauges, aircraft control and motion sensors and a recording system. These aircraft are then returned to service and used over the period of the programme in the normal mix of missions and flight environments. Typically the programmes gather data over a one to two year period.

The aims of the programmes are threefold:

(a) to establish the fatigue lives of major components through direct load measurement and thus ensure that promulgated fatigue lives are appropriate for the usage of the aircraft,

(b) to put a structural cost on manoeuvres/operations to permit cost effective management of the fleet,

and (c) to determine an efficient means of monitoring the structural usage of the aircraft or aircraft component.

Since the 1950s the structural usage of fixed wing aircraft has been monitored on an individual aircraft basis using information pertinent to the loading actually experienced by the airframe during a mission. For the helicopter, fatigue lives are usually quoted in terms of flying hours. From a logistic point of view, fatigue monitoring is simple but many questionable assumptions have been invoked.

In calculating fatigue lives for helicopter components, it is necessary to describe how the helicopter will be used in terms of:

(i) configuration details such as weight, cg and if applicable cargo type, etc.

and (ii) flight conditions such as hovering, transitions to and from the hover, ascent, descent, level flight, left and right turns, autorotation, spot turns, backwards and sidewards flight, etc.

When the severity of each category is taken into account, it is not uncommon for several thousand separate conditions to be admitted to the calculation. For each such usage condition, it is necessary to deduce, by calculation or flight test, the fatigue damage attributable to unit time in that condition. This has to be done for each component under consideration. Fatigue lives for the assumed usage can then be evaluated. The accuracy of the calculated fatigue lives depends upon the appropriateness of the designated flight conditions to cover the fatigue damaging activities in the military operational usage and the adequacy of the corresponding damage estimates derived as they are from loads or stresses measured during flight test often during the development phase. There is thus frequently a need to distinguish between the different roles in which the helicopter is deployed and to recognize that changes of tactics, etc within a role can have a marked effect on loads or stresses generated with consequential effect on fatigue damage. Clearly an operational assessment of the loading actions on helicopters is essential if realistic fatigue lives of components are to be forthcoming.

2 QUANTIFICATION OF HELICOPTER USAGE IN THE UK

The need for information on the way helicopters are used was recognized in the early 1970s but both the recording technology and instrumentation techniques of the day

prohibited all but questionnaire type surveys. Towards the end of the decade, an analogue u-v recorder was used to record aircraft response parameters[1], necessitating time consuming manual analysis. By 1980, compact digital recording equipment had been developed[2] and was used in a one-off programme on an in-service Sea King[3]. This trial installation was to be followed by a five-aircraft programme covering the different marks of Sea King helicopter and role usages.

The data collection philosophy adopted followed that of the helicopter fatigue evaluation. The service aircraft were equipped to measure aircraft control and response data, collectively referred to as parametric data, from which the analyst attempted to identify specific manoeuvres and/or conditions. While the data of the trial installation on Sea King was of sufficient quality to permit automatic analysis, it proved difficult to recognise specific manoeuvres consistently. The structural implications of the parametric data were to be assessed by reference to a data base collected on a flight test vehicle which, in addition to the service instrumentation, was comprehensively strain gauged. From the latter data source, it was hoped to establish load/parameter relationships which could be used to interpret the operational parametric data. In the event, it was found that the number of and complex interrelation between the flight variables in manoeuvring flight, particularly at low speed, was likely to defeat attempts to deduce the associated stresses accurately. Some direct load measurement would therefore be necessary.

Financial restraints delayed the follow-up in-service measurement programme on Sea King. However this allowed the programme to be reconfigured to include direct load measurement. In addition to recording dynamic system and airframe fatigue related data, data pertinent to engine health and usage will be recorded. However with current slippages in the programme, instrumented aircraft are unlikely to be in-service before 1988. It is planned to follow the Sea King programme with a six-aircraft programme on Lynx. The programme on Chinook, originally scheduled downstream of the Sea King programme, included direct load measurement from the outset and is now operational with three instrumented aircraft in service.

In the meanwhile, the recording equipment of the trial installation on Sea King has been used in an <u>ad hoc</u> programme on a Sea King Mk III with some direct load measurement. These data together with that from the Chinook programme are used in this Report to illustrate the type of data analysis to be pursued and to highlight problems that must be addressed if comprehensive fatigue monitoring on helicopters is to be realised.

3 IMPLICATION OF DIRECT LOAD MEASUREMENT ON IN-SERVICE DATA ACQUISITION

The decision to include direct load measurements in addition to aircraft control and response parameters has repercussions for in-service data collection which are circumvented with a purely 'parametric' installation.

The need to acquire direct loading data from the dynamic system necessitates transfer of data from the rotating components to within the fuselage. This has been traditionally done through the use of slip ring assemblies. Although slip rings are now reliable enough to be used for flight test applications, the maintainability problems on an in-service aircraft programme are likely to be high for that environment. Nevertheless they are used in the Chinook installation and their record to date is good. To improve data reliability RF transmitter/receiver devices are being developed, and are included in monitoring equipments for WHL 30-300 and EH 101. A single channel device is included in the <u>ad hoc</u> Sea King programme and a five-channel variant is under development for the five-aircraft programme.

Virtually all OLM programmes utilise the Plessey HUMS data acquisition system or one of its derivatives[2]. The equipment is relatively compact and has been shown to produce data of sufficient quality to analyse automatically in diverse environments. A bench test layout is shown in Fig 1. Data are recorded on compact cassette which is very convenient for front line usage. Overall system data rates and corresponding tape durations are shown in the table below.

Overall sampling rate (samples/s)	Tape recording duration (hours)
128	4½
192	3⅜
256	2¼
512	1⅛

These data rates are low compared with those necessary to acquire loading or stress data from a helicopter dynamic system or indeed from airframe structure whose loads are a function of the higher orders of rotor speed. It is therefore necessary to restrict the number of load channels that are examined simultaneously or invoke some pre-conditioning or analysis of the raw loading signals prior to recording such that the required sampling rate is much reduced.

Fatigue significant loading cycles* in a load time history are identified by the range-mean-pairs (rainflow) counting technique[4] and are described by their amplitudes and mean values. Peak-trough pairs identified as loading cycles are not necessarily adjacent in the time history. Loading cycles whose amplitudes fall below the fatigue endurance limit can be discounted. If however the mean of the load does not vary then

the loading cycles are formed from adjacent peak-trough pairs. Since the fatigue relevant loading histories for helicopter components have approximately this characteristic, except when gross changes of condition are involved, a vibratory signal which is described in terms of two dc signals, one representing the local amplitude and the other the local mean load can provide data of usable accuracy. Moreover from a fatigue standpoint damage varies as the third or fourth power of the amplitude of the load cycle and is roughly linear with the mean. In some cases the effect of mean load variations is not sufficiently well defined or large enough to be worth including and the 'amplitude' signal can be used in its own right. When substantial mean load variations do occur between manoeuvre states, it is necessary to reconstitute a single load history from the mean and vibratory signals from which loading cycles, which comprise a peak and trough from different manoeuvres or conditions, may be identified. The frequency of occurrence of the underlying loading cycles must be supplied by separate data sources and introduced into the fatigue calculation. This is true whether or not recombination of the mean and amplitude signals is necessary.

The Chinook installation includes a microprocessor-based peak-to-peak detector unit which produces a dc signal proportional to the maximum range excursion of a load history over a number of rotor cycles. The five-aircraft Sea King programme likewise will invoke raw signal conditioning so that nine direct load measurements can be described within an overall sampling rate of 256 samples/s. In the ad hoc Sea King programme, a single load channel was sampled 72 times during a second, while control and response parameters and housekeeping data occupied the remaining 120 positions in the overall sampling rate of 192. The parameters and their sampling rates are given in Table 1 for the Chinook programmes and in Table 2 for the ad hoc Sea King programme. Table 3 contains the proposed parameters to be recorded in the five-aircraft Sea King programme.

4 IMPLEMENTATION OF OLM

The design and installation of the OLM instrumentation and the analysis of the results are usually carried out by the aircraft manufacturer under MOD contract. RAE provides technical monitoring of such programmes and is actively involved in the development of analysis techniques. The strain gauge installations are designed to establish the fatigue lives of critically lifed items and to monitor the fatigue loading actions in the major load paths. The direct load measurements are used to calculate fatigue damage for related structural features. During the course of the analysis, the results from direct load measurements will be compared with those generated from usage spectra. Occurrences and conditions leading to high rates of fatigue damage will be identified separately and reported.

Component stresses, during particular manoeuvres and/or flight conditions, and hence fatigue damage are markedly dependent on aircraft configuration, viz weight and cg position. Damage usually increases markedly at high weights. It may well be that configuration details are the most fundamental parameters to be tracked in a fatigue evaluation programme but they are, unfortunately, the least amenable to automatic recording. While changes due to fuel usage could be tracked through fuel flow, fuel capacity, etc changes in internal load must be supplied by the operator. External loads are tracked and recorded through hook load sensors. Fig 2 shows the marked effect of changing weight and cg position on the level of structural loading, eg Aft Shaft Torque (AST), when a 5000kg load is carried sequentially on the forward (FHL), centre (CHL) and aft (AHL) hooks of Chinook. Thus it can be seen that configuration data are vital if comparisons with design usage spectra are to be possible and also for the results to be of maximum benefit to the fleet manager. Visibility of the origins of fatigue is very important for the operator seeking to equalise fatigue consumption across the fleet. Accordingly in the Chinook programme, utilization and fatigue damage are to be logged against 625 usage categories which comprise five weight bands, five centre of gravity bands, five density altitude bands and five speed bands. Internal and external cargo configurations will be separately logged. From this information damage in the different roles will be compared.

Since in the OLM programme, read across to the other aircraft in the fleet will, in the first instance, be via the flight hours logged against the airframe, it is important that the definition of flight time used in the analysis of OLM data parallels that in the aircraft log. The need for care can be seen from a consideration of the way in which support helicopters are used. Here, for example, time on the ground for aircraft loading/unloading is usually included in mission flight time, whereas more prolonged stops would not.

Additionally occurrence spectra such as the number of rotor start/stops, landings and autorotations will be determined. Usage spectra such as time spent in predefined conditions, eg single engine operation, levels of engine torque and altitude/airspeed bands, will also be evaluated.

5 MANAGEMENT AND ANALYSIS OF OLM DATA

5.1 General philosophy

Although automatic processing of the operational data is essential, it is important to have visibility of the results in an easily digestible form. While it would be

* A loading cycle is formed when a load history rises through a particular level to a local peak, falls to a trough at the particular level and subsequently rises beyond the peak level value. The peak and trough values characterize the loading cycle.

impossible to examine all records in fine detail, it is necessary to ensure that structurally relevant information is not overlooked. The analysis techniques are designed to give the engineer as much detail as required in special investigations while being manageable by relatively junior staff.

In processing data from OLM programmes, it has been found necessary to introduce a cataloguing system to monitor progress of individual flights during the analysis process, and to provide a means of categorising each flight so that like flights may be grouped together and salient results pooled. Up to 30 separately identifiable characterisation or occurrence codes together with a numerical coding of mission type are available in the programme in use at RAE. In addition, summary programmes are used to collate and pool results from specific profiles which can be defined by mission type or character code either singly or in any combination. An example is shown in Fig 3 which compares normal g exceedance spectra for three different taskings A, B, and C with that obtained overall. Although the versatility of the helicopter can make categorization difficult beyond the theatre of operation, such pooled results can be useful for fleet management and design purposes. This fine detail can only be supplied by the operator and is not normally produced for any record purposes. Therefore if such information is to be forthcoming, supporting documentation is required from operational units.

5.2 Interpretation of direct load measurements

The fatigue damages calculated from the direct load measurement are evaluated on a flight-by-flight basis. In the analysis programs developed at RAE, results from flights belonging to a specific profile are grouped together and statistics on constituent flights produced to provide an overview. The statistics include data of relevance to static strength considerations so that important load level exceedances can be isolated and investigated. The overview contains:

(i) a list of flight numbers of the set,

(ii) for each flight, the maximum and minimum of each load channel and corresponding time into flight. (This item is optional.)

(iii) the flight number and time into flight of the maximum and minimum of each load channel over the profile,

(iv) significant deviations of the maximum and minimum of each load channel compared with the average over the profile,

(v) fatigue damage statistics over all flights of the profile for all channels,

and (vi) graphical presentation comparing the average damage with that of individual flights for user-selected channels.

By virtue of (vi) extremes of fatigue damage within the set can be identified and, through the detailed results of particular flights, studied in depth. During analysis, damage accumulation profiles[5] are produced which enable high local damage rates to be identified. A typical example is shown in Fig 4a which depicts the damage growth through the flight for a lift frame feature when held up against one of the S-N curves under investigation. This is to be viewed in conjunction with the coarse flight profile information of Fig 4b which shows the cumulative count of the number of air-ground and ground-air transitions, (in which an airborne state corresponds to an odd number and a ground state to an even one) and in shaded blocked format, the ranges of rotor RPM, airspeed and altitude over an interval of time. By comparing Figs 4a&b, it can be seen that the high rates of damage occurring between 0-10 minutes and 70-80 minutes into the flight are associated with air-ground-air transitions. Clearly, as far as the lift frame is concerned, the ground-air-ground cycles are a major source of fatigue loading. This is confirmed in Fig 5 which shows the location of fatigue damaging loading cycles and their relative fatigue damage contributions*, as a function of their amplitude and mean. The largest ground-air-ground cycle contained in the flight of Fig 5 is some 27 units of micro damage. However, when results from a number of flights are examined, it is evident that ground-air-ground damage can vary by a factor of more than 2 in either direction. Among the factors which contribute to this variation are: differences in configuration, ie weight etc; landing loads; the occurrence of turbulence; manoeuvring; and the severity of the approach to hover. The latter is illustrated in Fig 6a&b which respectively illustrate a repositioning flight with substantially no forward speed and one which involves a transition to the hover. The amplitude and duration of the increased activity on the lift frame in the highlighted area A of Fig 6b is associated with rate of change of torque. The relationship is not a simple one. In general terms the larger amplitudes are generated above a certain rate of change of torque but the longer durations are usually associated with the more moderate rates.

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* For a particular structural feature, the fatigue safe life is reached after the application of a number of loading cycles which may differ in severity. The total amount of damage sustainable by a component is given the value unity, and under the Miner-Palmgren hypothesis, each loading cycle contributes independently to the accumulation of fatigue damage. The amount of damage done by individual cycles is small and it is convenient to work in units of 'damage $\times 10^{-6}$' so that the safe life of the component is reached at 10^6 units.

The damage during the airborne phase also covers a broad range in damage terms, the larger amplitude cycles being associated with, or between, manoeuvres which involve large engine torque variations. Typical strain gauge signatures are shown in Fig 7. Also shown are responses obtained in turbulent conditions which include low frequency modulation of the normal signature.

To provide further visibility of the origins of fatigue damaging regimes, it was suggested in section 4 that damage and operating time be split up into various usage bands. The supporting weight information in this programme was not of sufficient quality to attempt any breakdown with respect to weight, but that with height and speed is shown graphically in Fig 8a. Thus for this sortie 50% of the time was spent in the band "altitude <1000 ft and speed >100 kn", but it represented only 8% of the fatigue damage. On the other hand some 70% of the damage occurs when the speed is less than 40 kn. Other sortie types produce different distributions of operating time and associated damage with respect to altitude/airspeed bands, eg Fig 8b.

5.3 Effect of grouping loading cycle data

As is evident from Fig 5, the fatigue damage due to a loading cycle is a function of the amplitude and mean of the cycle. This function is frequently described algebraically. During the course of the analysis described above, loading cycles were grouped and counted into cells of an amplitude-mean frequency of occurrence matrix. All counts in an element of such a matrix are attributed to the amplitude and mean values appropriate to the centroid of the cell. This accounts for the discrete positioning of the location of the loading cycles shown in Fig 5. The damage for the flight was evaluated as 122.5 which is comparable with 120.4 calculated by treating each loading cycle individually. Differences of this magnitude, representing some 2-3% of the total, were typical. This result is of particular importance for on-board monitoring systems wherein fatigue endurance performance may not be fully established prior to service usage. Reinterpretation of loading cycle data for other S-N data should be possible from a frequency of occurrence matrix.

6 INSTRUMENTATION INTEGRITY AND ON-BOARD ANALYSIS

The Sea King installation utilises a FM telemetry torquemeter and signal transmission system and thus avoids the use of slip ring assemblies to gather data from the rotating dynamic system. Although the system appeared to function well for considerable periods, there were occasional spikes in the waveform such as that highlighted in Fig 6b (area B) which are considered to be erroneous since they are not supported by corresponding motion on other measurands. The signature of the erroneous signals do not differ sufficiently from other seemingly valid excursions for error recognition techniques to isolate them. Reasons for their occurrence are being investigated. These and other instrumentation faults which ostensibly cannot be isolated by simple data validation algorithms have important ramifications for on-board processing of such data. Analysis algorithms must be constructed which permit result validation after analysis.

Interference from high frequency radiation sources can also destroy the quality of a measurand. Freedom from mutual interference problems is an essential ingredient for any on-board analysis system. During the initial flight testing of the Sea King installation, it was found that HF radio usage interfered with the instrumentation. While decoupling capacitors improved the situation, as can be seen by Fig 9, it was still necessary to mark periods of HF activity and omit them from the analysis. Improved screening techniques should improve the situation further. However, before airborne analysis is attempted, it is suggested that stringent checks must be carried out on the installation using raw data recording and on-ground examination of the data.

7 CONCLUSIONS

Operational Load Measurement programmes on helicopters provide a means of substantiating the fatigue lives of safe-life components promulgated on the basis of assumed operating spectra. In addition usage spectra and fatigue damage can be related to specific roles and mission types. Thus the programmes offer the operator accurate information on which to base fleet management from a structural standpoint.

In the UK, Operational Load Measurement programmes are in progress on Chinook and programmes on Sea King and Lynx are planned. The programmes are considered to be the only reliable way of quantifying fatigue usage and establishing accurate fatigue lives.

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4 Susan D. Ellis. A combined range-mean-pairs rainflow count of load time histories for use in the formulation of structurally relevant cost functions and fatigue analysis. RAE Technical Report 81122 (1981).

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Table 1

CHINOOK HC MK 1 PROGRAMME: RECORDED PARAMETERS, IDENTIFICATION CODES AND SAMPLING RATES

Identification Code	Parameter	Sampling rate
RPM	Rotor speed	2
IAS	Airspeed	2
OAT	Outside air temperature	1
ALT	Altitude	2
LOC	Longitudinal stick	8
LAC	Lateral stick	8
RUD	Pedal	8
COL	Collective pitch	8
PCH	Pitch attitude	8
ROL	Roll attitude	8
HDG	Heading	8
FHL	Forward hook load	1
CHL	Centre hook load	1
AHL	Aft hook load	1
NZG	cg vertical acceleration	8
NYG	cg lateral acceleration	4
FFL	Forward fixed link (dc signal)	4 Direct load measurement
AFL	Aft fixed link (dc signal)	4 Direct load measurement
AB1	Aft shaft bending No.1 (dc signal)	2 Direct load measurement
AB2	Aft shaft bending No.2 (dc signal)	2 Direct load measurement
AST	Aft shaft torque	4 Direct load measurement
RAD	Radar altimeter	2
CGI	CGI signal	2
TQ1	No.1 engine torque	2
TQ2	No.2 engine torque	2
PT1	No.1 engine PTIT	2
PT2	No.2 engine PTIT	2
N11	No.1 engine gas producer speed (N1)	2
N12	No.2 engine gas producer speed (N1)	2
N21	No.1 engine power turbine speed (N2)	2
N22	No.2 engine power turbine speed (N2)	2
P31	No.1 engine P3	2
P32	No.2 engine P3	2
	Discretes:	2
	Squat switch	
	AFCS switch position - 1,2, both, off	
	Engine 1 or 2 anti-ice switch - on/off	
	Barometric altitude hold - engage/off	
	Radar altitude hold - engage/off	
	Crewman station operated on/off switch	
	Longitudinal cyclic trim switch - manual/auto	
	Dash actuator failure switch - operating/failed	
	HF markers	

Table 2

AD HOC SEA KING PROGRAMME: RECORDED PARMETERS, IDENTIFICATION CODES AND SAM LING RATES

Identification Code	Parameter	Sampling rate
IAS	Airspeed	3
ALT	Altitude	3
OAT	Outside air temperature (not available)	3
PCH	Pitch attitude	6
ROL	Roll attitude	6
HDG	Heading	3
TQ1	No.1 engine torque	6
TQ2	No.2 engine torque	6
NZP	Normal G rotor centre line - port	6
NSZ	Normal G rotor centre line - starboard	6
NZA	Normal G aircraft centre line - aft	6
NX	Longitudinal G rotor and aircraft centre line	3
NYF	Lateral G forward	6
NYA	Lateral G Aft	6
COL	Collective pitch position	3
LAC	Lateral cyclic stick position	6
LOC	Longitudinal cyclic stick position	6
YAW	Yaw control position	6
RPM	Main rotor speed	3
TRT	Tail drive shaft torque	6
SG1	Strain gauge on main lift frame starboard side aft	72 Direct load measurement
BCL	Tail rotor bell crank lever load	6
UNL	Underslung load (not available)	6
	Discretes:	3
	Load event	
	AFCS	
	Squat switch	
	HF markers	

Table 3

PARAMETERS TO BE RECORDED IN FIVE AIRCRAFT SEA KING PROGRAMME

Pressure altitude
Indicated airspeed
Outside air temperature
Pitch attitude
Roll attitude
Heading
Main rotor RPM
No.1 engine torque
No.2 engine torque
Aircraft normal acceleration rotor centre line port
Aircraft normal acceleration rotor centre line starboard
Aircraft normal acceleration aft
Aircraft fore/aft acceleration
Aircraft lateral acceleration forward
Aircraft lateral acceleration aft
Collective position
Cyclic lateral position
Cyclic longitudinal position
Yaw pedal position
Collective servo position
Cyclic lateral servo position
Cyclic longitudinal servo position
Yaw servo position
Tail drive shaft torque

Direct load measurements	- main lift frame station forward port
	- main lift frame station staboard aft
	- tail pylon hinge
	- tail rotor bellcrank lever
	- tail rotor spider arm
	- main rotor rotating star arm
	- main rotor shaft upper
	- main rotor shaft lower
	- main rotor top hub plate

Underslung load
Main gearbox oil temperature

Power turbine inlet temperature	For both engines
Fuel flow rate	
Fuel temperature	
Gas generator speed	
Power turbine speed	
Compressor delivery air temperature	
Compressor delivery air pressure	
Inlet guide vane angle	

Discretes: AFCS event
Oleo switch event
Load event

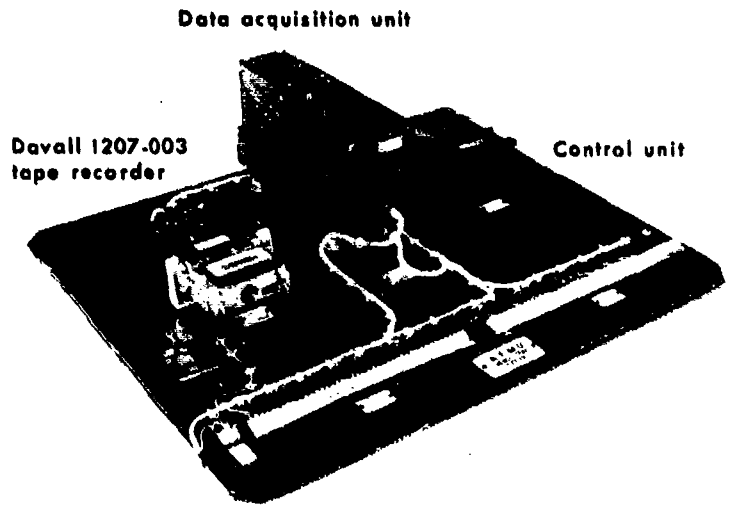


Fig 1 Operational load data acquisition system: bench test layout

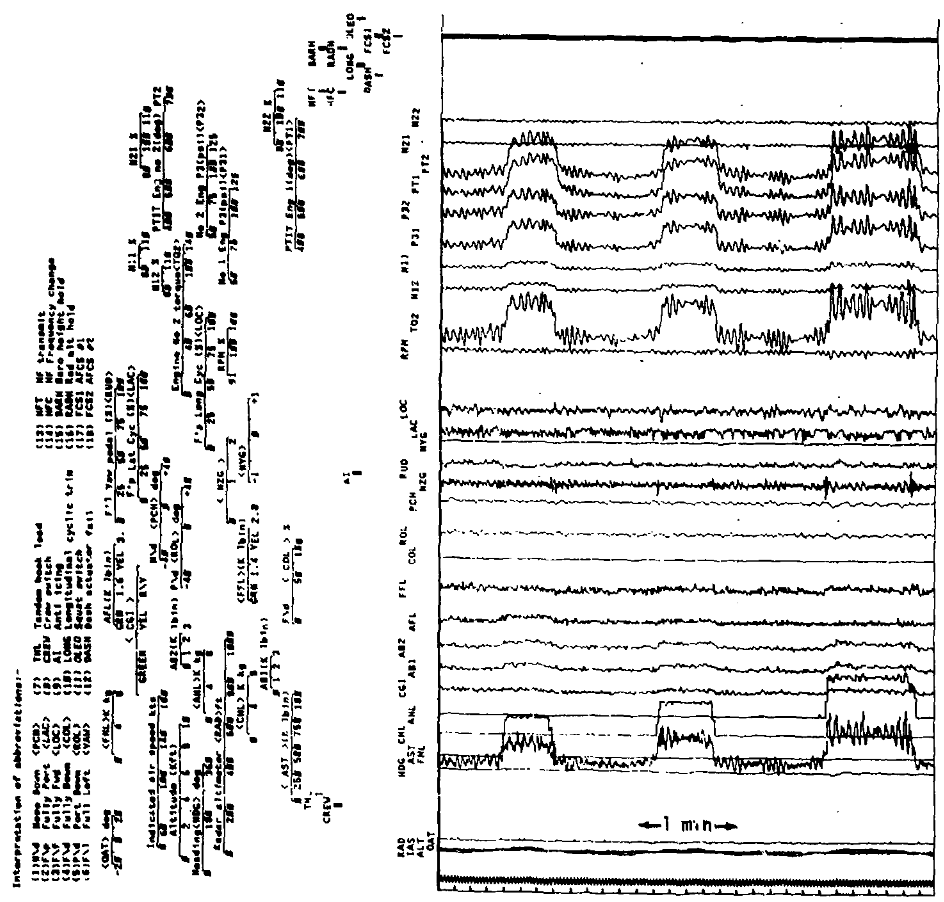


Fig 2 Chinook external load calibration trials: Effect of weight and cg position on load histories

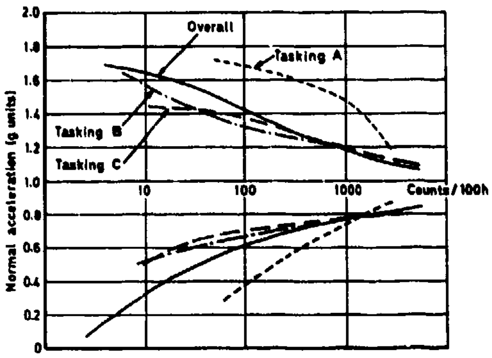


Fig 3 Comparison of normal acceleration exceedance curves for different taskings

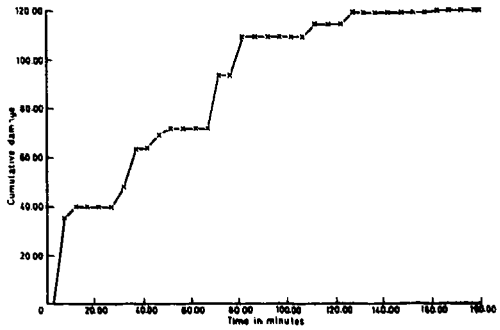


Fig 4a Damage accumulation profile over flight time: Flight 1

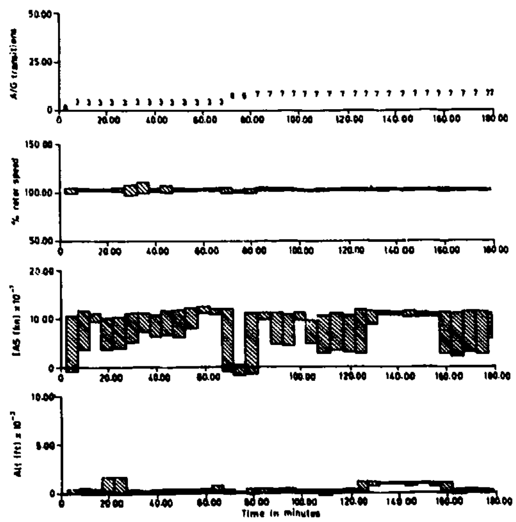


Fig 4b Supporting parameters for damage accumulation profile: Flight 1

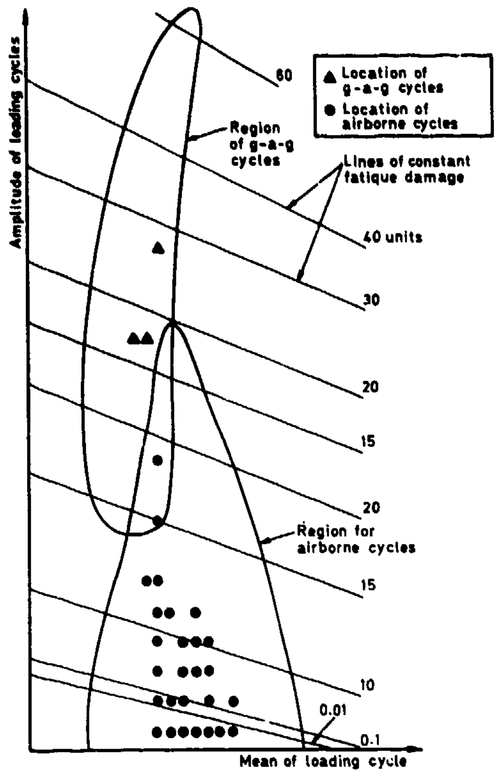


Fig 5 Location of loading cycles with S-N data plotted as constant damage curves: Flight 1

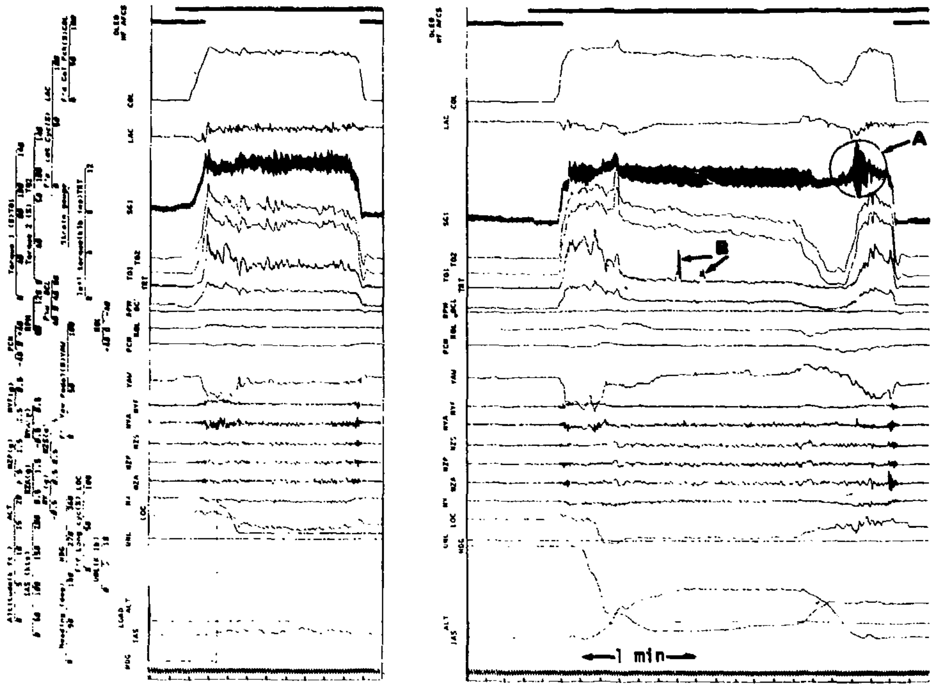


Fig 6a Repositioning manoeuvre

Fig 6b 3 minute flight: effect of transition to hover on lift frame strain gauge (area A)

Fig 6 Examples of data collected during OLM programmes: Sea King

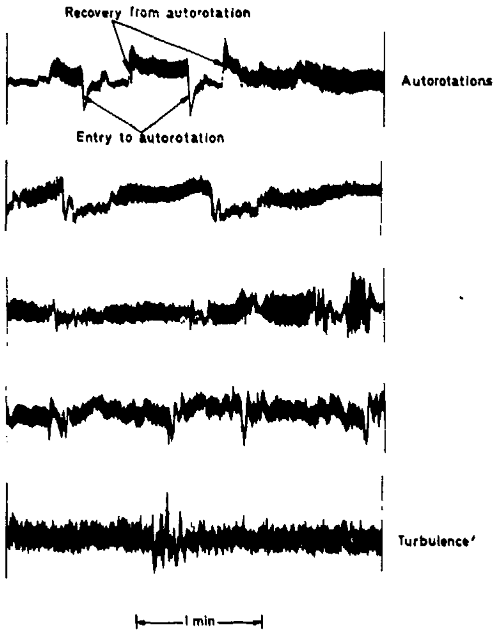
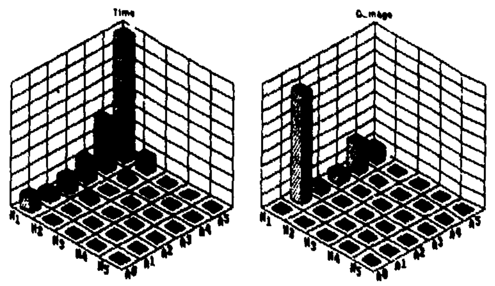
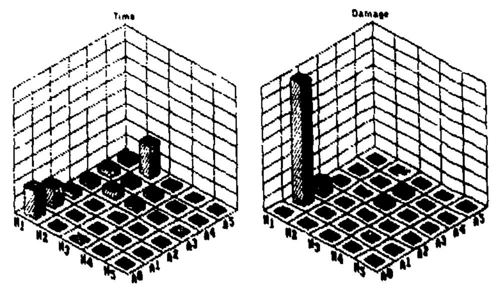


Fig 7 Lift frame strain gauge signal obtained in flight: typical variations due to torque changes and turbulence



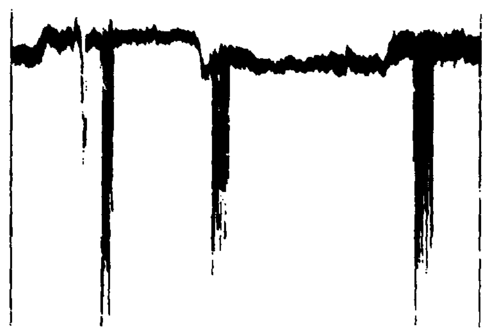
Altitude bands <1000ft → H1, 1000ft - 2000ft → H2, 2000ft - 3000ft → H3, 3000ft - 4000ft → H4, >5000ft → H5
Airspeed bands ground → A0, <40kn → A1, 40kn - 60kn → A2, 60kn - 80kn → A3, 80kn - 100kn → A4, >100kn → A5

Fig 8a Distribution of time spent and fatigue damage with respect to altitude/airspeed bands: Flight 1



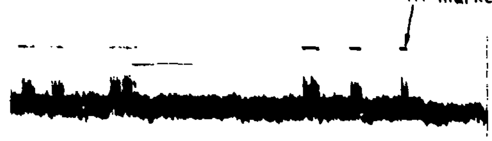
Altitude bands <1000ft → H1, 1000ft - 2000ft → H2, 2000ft - 3000ft → H3, 3000ft - 4000ft → H4, >5000ft → H5
Airspeed bands ground → A0, <40kn → A1, 40kn - 60kn → A2, 60kn - 80kn → A3, 80kn - 100kn → A4, >100kn → A5

Fig 8b Distribution of time spent and fatigue damage with respect to altitude/airspeed bands: Flight 2



a Original installation

HF marker



b Installation after modification

|— 1 min —|

Fig 9 Effect of HF radio transmission on strain gauge signals

THE FLIGHT EVALUATION OF AN ADVANCED ENGINE DISPLAY AND MONITORING SYSTEM

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SUMMARY

A Wessex helicopter at RAE Bedford has been used to develop and evaluate a suite of advanced, integrated avionics. An important area of study has been concerned with the display of engine and transmission data, and with a system monitor which gives audio and visual warning of any problems.

The system has demonstrated that the suppression of engine and transmission data at all times except when the pilot asks for it to be displayed or the system detects a problem is an acceptable technique. The use of synthetic voice output has meant that the pilot can spend a greater proportion of his time looking out of the aircraft; the additional head-out time enables mission effectiveness and flight safety to be enhanced.

1 INTRODUCTION

A Wessex helicopter at RAE Bedford is being used to evaluate and demonstrate an integrated cockpit and flight management system. The equipment installed includes colour displays, a monochrome control and display unit, a speech recogniser, synthetic speech output, and interfaces to most of the aircraft systems and instrumentation. The hardware and software have been carefully designed to form an easy to use, pilot friendly system.

This paper describes the operation of the system, with particular emphasis placed on the engine monitoring and display aspects. Throughout the programme, attention has been concentrated on the systems integration aspects of harnessing the new technologies that are now becoming available to the avionics engineer. This paper attempts to show how this can be achieved with the engine monitoring and display system.

2 WESSEX FACILITIES

A standard Wessex 2 was modified to incorporate a suite of advanced, integrated avionics. This involved removing all electromechanical instruments from the left-hand instrument panel, and installing the following displays and controls in their place:

- Two colour displays, one with a touch sensitive overlay
- A monochrome Control and Display Unit (CDU), with touch sensitive overlay and associated hard keys
- 16 key keypad
- Speech recogniser
- Synthetic voice output
- Cursor controls and voice recogniser activation switch on the cyclic control
- Collective mounted joystick control for use with the map.

The right-hand instrument panel was rearranged to accommodate a colour display in addition to the electromechanical instruments, so that the safety pilot or observer in the right-hand seat could view either of the two colour display formats which were being presented to the subject pilot in the left-hand seat. Fig 1 is a photograph of the re-equipped Wessex cockpit.

The system consisted of three main processors, which were interfaced to each other, to the display generators, and to the following aircraft systems:

- Airspeed
- Barometric height
- Radio height
- Doppler groundspeed
- Outside air temperature
- Heading, pitch and roll attitude
- Normal and lateral acceleration
- Collective and cyclic flying control positions
- Instrument Landing System (ILS)
- Microwave Landing System (MLS)
- Microwave Aircraft Digital Guidance Equipment (MADGE)
- Engine and transmission data:

Power turbine speeds
Compressor speeds
Power turbine inlet temperatures
Fuel flows
Fuel contents
Rotor rpm
Torque

Digitally controlled UHF/VHF radio
Digital map
Speech recogniser
Synthetic speech output
Radio clock

There were also facilities for recording any of the video displays onto tape, and also for recording any required flight parameter for later replay and analysis.

3 COLOUR DISPLAYS

The displays used were commercially produced, general purpose domestic monitors, which were repackaged for airborne use. They were fed with a PAL video signal from their signal generators via an electronic switching unit, which allowed either of the two display formats to be displayed on either monitor. The signal generators were capable of producing eight colours, with a pixel resolution of 512x512.

One of the displays contained all information required by the pilot to fly the aircraft, and this was called the Primary Flight Display (PFD). The instruments on this display enabled the pilot to fly in all modes of flight, and presented him with any required guidance information.

The other display was called the Multi-Function Display (MFD), and was used to present all other information that the pilot might require. Usually the navigation format was displayed, and this was formed by overlaying navigation and route information on a digital map.

Many other formats could be displayed, either at the request of the pilot (by using the CDU or speech recogniser), or automatically when the system detected a problem about which the pilot needed to be warned. The engine display was one of these, and this was selected by the pilot during engine start-up, shutdown, and whenever he wished to check his engine instruments. As the system continuously monitored all parameters, the Temperatures and Pressures (T's and P's) format was suppressed for the majority of the time, and the MFD was used for other purposes.

If the system detected that one of the monitored parameters was out of limits, the situation was flagged to the pilot in up to three ways. In most cases the problem was presented on the CDU, the MFD changed its display format to give a display relevant to the fault, together with a check-list of actions required, and the synthetic voice output said the word 'warning', followed by a description of the problem. Warnings such as torque exceedances, where the pilot needed no further information, were flagged using voice only. Warnings related to data on the PFD (which was presented to the pilot all of the time) were flagged using voice and a change in the CDU display.

3.1 <u>The T's and P's page</u>

The objective when developing a format for the display of engine data was to produce a clear, uncluttered display, which could be used in all phases of flight, and would reduce the pilot's workload.

The information that was shown on the display could be split into two groups:

(i) Port and starboard compressor speeds (Ng)
Port and starboard free power turbine speeds (Nf)
Port and starboard power turbine inlet temperatures (T4)
Port and starboard fuel flows
Port and starboard fuel contents
Rotor rpm (Nr)
Rotor torque

(ii) Port and starboard hydraulic pressures
Port and starboard engine oil temperatures
Port and starboard engine oil pressures
Coupling gearbox oil temperature and pressure

The parameters in group (i) were all monitored by the microprocessor system continuously. However, during the current phase of the trials programme, the parameters in group (ii) were not coupled to the microprocessor system. The displays for these latter parameters were artificially generated using values that corresponded to cruise flight, so that an assessment of the entire display could be made.

The engine format adopted at the start of the programme is shown in Fig 2. Its main features were:

(i) Strip scales were used for all engine parameters; these allowed a reasonable packing density of information, without producing a cluttered display, and allowed trend information to be derived from them rapidly;

(ii) nonlinear scaling was used to give more movement in critical areas of operation;

(iii) the strip scales were complemented with digital readouts;

(iv) the parameters were grouped into the two engines separated by the rotor rpm, instead of the more usual convention of grouping by parameter. The object of this was to try to provide quicker diagnosis of engine problems;

(v) the strip displays for parameters were colour coded depending upon their value:

(a) if an engine parameter was within limits it was coloured white;

(b) if the rotor rpm was within limits it was coloured green - this gave a good dividing line between the port and starboard parameters;

(c) if any parameter was outside its limits it was drawn magenta.

(vi) although both rotor rpm and torque were displayed on the PFD, they were also presented on the T's and P's page. On the T's and P's page, the rotor rpm was displayed as a strip display, and the torque as a solid circle, a complete circle indicating maximum allowed torque. Any exceedance past this value resulted in the circle being redrawn in magenta.

After many hours of flying, the subject pilots pinpointed the following problem areas with this particular display format:

(i) There was no indication of the margin allowed on the parameters. There was good indication when a parameter was out of limits, but during particular situations, such as engine starts, it was useful to know when a parameter was about to go out of limits, so that avoiding action could be taken;

(ii) the Nr and Nf displays were too cluttered, and the lack of a digital readout for Nr produced complaints;

(iii) the T4 temperature strips were too small;

(iv) the digital readouts that existed were useful, but were wasteful of space. They were placed on top of the strips, moving as the parameters varied. During engine starts, this movement was too rapid, making the digits difficult to read;

(v) the relative positions of fuel flow and contents were poor. Because these values needed to be compared and balanced, it was thought better to place them adjacent to each other.

These shortcomings were overcome with the next version of the display, which is shown in Fig 3. On this display the digits were placed below the strips, so that they didn't move, and a digital readout of Nr was included. The fuel flows and contents were placed together in the lower corner of the display, together with an analogue balancing bar, which disappeared to nothing when the fuel flows were matched.

To give an indication of margin, a pair of horizontal lines were drawn across the strip displays. For a parameter to be within limits, its strip scale had to remain within these lines, which became known as the tram lines. The use of nonlinear scaling enabled the margins for all parameters to be indicated by a single pair of tram lines.

Simulated values for the parameters for which there were no sensors (ie the hydraulic and oil temperatures and pressures) were presented in a form similar to that for the parameters which were sensor derived. This enabled an assessment of a complete engine format to be made.

4 SYSTEM MONITORING AND WARNINGS

As mentioned above, the system monitored all the on-line parameters, and if a limit exceedance occurred the pilot was warned. The warnings were prioritised, and the way that the warnings were given to the pilot depended upon the severity of the problem.

There were basically three ways of giving a warning:

(i) Audio - an audio warning was given using the synthetic speech output system. This was found to be the most effective technique for bringing a problem to the attention of the pilot, who might be flying head-out for the majority of the time. Of course, a simple bell or klaxon could be effective at attracting the pilots attention, but it would give no information about the precise nature of the problem.

Using the speech output system, the warning was typically made up of two parts: the word 'warning', and a phrase indicating the nature of the problem, for example, 'check port T4'.

The torque warning, and other calls such as those for low height, were not preceded by the word 'warning'. The rationale behind this was that a rapid response was required, and that no further information was required form the system before the pilot reacted. The single word 'torque' was sufficient to indicate that the rotor was being over torqued, and pilots quickly reduced power. For all problems related to engines, however, the engine display had to be examined. In this case, the word 'warning' acted as a trigger for the pilot to look into the cockpit and at the engine display. If an audio warning was ignored, it was repeated every seven seconds. This length of time was sufficiently long to avoid nuisance warnings, but kept reminding the pilot that he had a problem with the aircraft, or was flying outside of its operating limits.

(ii) CDU - with most warnings, the CDU display was cleared, and a line of text indicating the reason for the warning was displayed. For example, following a power turbine inlet temperature exceedance, the line of text might be:

PORT PTIT > 710

The exact temperature displayed would depend upon the limits applicable to the current configuration of engines (one running, two running etc). The limits for all parameters varied in this way and were continuously updated by the flight management system. For example, the torque limit was 3000 lb/ft for two engines, 2700 lb/ft for one engine.

Displaying a warning message on the CDU effectively froze it until the warning had been acknowledged, either by touching the screen or by pressing the CDU reset button.

(iii) MFD - with engine parameter limit exceedances, the multifunction display format was changed, and a display relevant to the problem was presented to the pilot automatically. The display showed the type of failure (eg single engine), possible causes of the failure, a check-list of actions to be taken (replacing the manual flipcards), and a display of engine parameters. Thus on the single display the pilot had all the information needed to recover the engine. Against the check-list was a cursor, which he could move down the list as he completed the actions, using a button on the cyclic flying control.

With any form of automatic monitoring and warning system, nuisance warnings must be minimised, firstly because they distract the pilot, and secondly because they undermine his confidence in the system. The result of too many nuisance warnings is that the pilot reverts to performing conventional engine monitoring, which increases his workload and reduces mission effectiveness. The avoidance of nuisance warnings was partly achieved by inhibiting sub-sets of warnings, should a higher priority problem occur. Advisory messages, such as height calls when below 250 ft are not always required, and so could be turned on and off using a dedicated area at the bottom of the CDU. Turning the voice output on and off in this way caused it to say 'hello' and 'goodbye' respectively. Any transition through 500 ft caused it to come on and say 'hello'. This doubled as a confidence check, ensuring that the system was working.

4.1 Engine performance monitoring

The availability of altitude, pressure, outside air temperature and the engine parameters facilitated the on-line calculation of the Power Performance Indices (PPI) for each engine. The Wessex has a single torquemeter for the two engines, with the consequence that any PPI calculations have to be made with one engine pulled back to ground idle. To prevent nuisance warnings when pulling back the engine, all engine warnings were suppressed when the PPI page was selected. On more modern helicopters this will not be a problem, as each engine has its own torquemeter. This, together with a good engine model, will make on-line PPI monitoring and recording a viable proposition. General monitoring could be used to warn the pilot through the CDU or MFD of any reduction in engine performance; recording the data for later use by servicing crews could enable an engine's life to be directly related to its history, instead of just running time. Thus an engine that is cycled infrequently will have a longer life than one that is cycled often, reducing running costs and increasing reliability.

4.2 Aircraft performance

The performance of the aircraft as well as the engines was monitored, and its capabilities calculated in real time. Data from the operating data manual reference curves was incorporated into the system. This enabled the pilot to be informed of his hover capability, maximum airspeed etc, with the current configuration of all up weight and fuel remaining. The data could be accessed at any time through the flight status page of the MFD, where the all up weight, maximum airspeed, torque required for various forms of hover, and other parameters were presented in a tabular form.

To avoid the necessity of calling up a different page when in a critical stage of flight, such as during an engine failure, the hover capability could be displayed on the

T's and P's page. The way that this was operated in practice was that the pilot would pull in as much power as he could until one of the engine parameters came up to a limit, and then press one of the cursor buttons on the cyclic control. The system would then display the aircraft's hover capability using the maximum power available with one of the following messages:

HOVER OGE + TM	(hover out of ground effect with 5% thrust margin)
HOVER OGE	(hover out of ground effect with no margin)
10' HOVER	(10 ft hover capability)
0' HOVER	(0 ft hover capability)
NO HOVER CAPABILITY	

By having more accurate knowledge of the aircraft's capabilities at any stage during the flight, the pilot was able to fly closer to the edge of the performance envelope for more of the time.

5 FLIGHT EVALUATION

5.1 Engine starts

There are two situations during flight when the pilot needs to get information from his engine instruments quickly. These are the start up sequence and during engine failure.

When pilots first encountered the strip gauges in the aircraft there was no display present, as all parameters were zero. This was initially disconcerting, but once a pilot had experienced an engine start, there was no problem. Both trend and situation information could be determined from the format easily. This was demonstrated a number of times during the programme when hot and fast starts were experienced. On these occasions, detection of the fault was as fast or faster than when using mechanical instruments.

During engine starts the first problem that was encountered was a power turbine inlet temperature (T4) exceedance. This was easily picked up using the strip displays with the associated colour change but, if missed was picked up by the monitoring system and a warning issued.

The compressor speed (Ng) display gave plenty of information to allow the high pressure cock to be opened at the right time, and ground idle conditions to be set.

Part of the normal start-up sequence is to pull back the speed select lever of each engine, to check the correct operation of the freewheel. This triggered some of the single engine failure warnings, and acted as a confidence check on the system.

Once both engines were running, the balancing bar on the fuel flow proved to be a useful aid, enabling the fuel flows of the two engines to be matched very easily.

5.2 In flight monitoring

Flight evaluation of the engine display and monitoring system was made during all trial sorties. (The total flying hours for the programme was approximately two hundred and fifty.) Engine monitoring was therefore evaluated as part of a complete system, and not as a separate entity. During normal flight, the navigation display, or a status page such as a radio channel plan or waypoint list was displayed on the MFD. Depending upon the tasks that he had to perform, the pilot would select the T's and P's format from time to time. With the monitoring system functioning, there was in fact no necessity for the pilot to check the display. However he did so to establish confidence in the system.

During the trials, although not all warnings were triggered, a large number were tested, either deliberately or accidentally.

In normal flight, it was not uncommon for slight torque exceedances to occur, or for the aircraft's maximum speed (Vmax) to be exceeded. When this happened, the voice warning was usually enough to alert the pilot to the problem, especially with the torque warning. Usually the pilot did not even bother to bring his head into the cockpit to look at the display, though if he was flying head down the colour change on the PFD alerted him to the problem.

Apart from artificially generated fuel flow problems, there were a number of flights when the pilot's fuel management was sub-standard. In these circumstances genuine warnings of fuel flow mismatch, fuel contents mismatch, and low fuel occurred. These warnings were effective and the pilot was rapidly made aware of the problem and was able to take corrective action.

Although no actual engine failures occurred during the trial, single engine failures were often simulated by pulling back one of the speed select levers. Obviously not all modes of failure could be simulated (for example runaways), but those that could were used to prove the concepts of the warning system and to demonstrate that the proper corrective action was prompted from the pilot.

During a single engine failure, the pilots praised the ability to calculate the hover capability of the aircraft instantly. To determine what hover capability the aircraft had, the pilot simply pulled in power until one of his engine parameters, such as

a T4, came into limit. He then pressed one of the cursor control buttons on the cyclic control. His hover capability was then instantly displayed on the T's and P's page.

5.3 Checks

The ability to call up checks associated with the T's and P's format was found to be very useful. For example, the pilot could display both his take off checks and the engine parameters together on the same screen at the same time, as shown in Fig 4. This combination proved very useful both before and after take off. The combined display not only reduced the number of selections that the pilot had to make through the CDU, but enabled him to view his primary flying instruments, the check-list of interest, and the engine instruments.

Similarly, when dealing with a problem such as an engine restart in flight, all the data and the check-lists would be presented to the pilot together on the one display.

6 T'S AND P'S FORMAT

With conventional electromechanical instruments, gauges are grouped according to parameter type. For instance, the two T4s will be placed next to each other, as will the two Nrs, etc. This sometimes causes problems when trying to isolate a problem. It is not unknown for a pilot to pull back a good, working engine, because he has misinterpreted his instruments. To try and overcome this problem, the parameters on the electronic display were separated into the two engines, and displayed in a symmetrical pattern about the rotor rpm strip gauge, as shown in Fig 5. Any failure of an engine will result in an assymetric pattern, both in colour and in shape, and the faulty engine can quickly be identified.

Another important feature of the format was the introduction of tram lines running across it, representing the normal operating limits of the parameters. With the aircraft operating normally, all parameters will lie between the two tram lines.

To achieve tram lines that ran straight across all parameters, nonlinear scaling was adopted. Using tram lines meant that not only was fault diagnosis easier, but so too were power checking and engine condition monitoring, such as calculating Power Performance Indices (PPIs). For example, during take off, the pilot only had to scan the T's and P's format quickly to ensure that all parameters were within the tram lines; he did not have to make a complete scan of a large number of different gauges.

The limits on the display were automatically configured to reflect the condition of the engines. For instance, the torque limit changed from 3000 lb/ft to 2700 lb/ft when only one engine was running.

7 PRIMARY FLIGHT DISPLAY

The PFD was used to present to the pilot all of the basic flying instruments, and was displayed at all times. Torque and rotor speed were included on this display, torque because it must be referred to constantly, and rotor speed because it is required together with attitude information during emergency procedures such as an auto rotation. Fig 6 is an example of the PFD format.

When the rotor is operating normally and within limits, the display simply consists of an unobtrusive white bar. As the rotor speed moves out of limit, a coloured strip begins to move out of the white bar, to the right if the speed is above normal, and to the left if it is below. Eventually the strip is replaced by digits, as a second set of limits is reached. This second set of limits was chosen to provide a very responsive display over the area available. The presentation of data in digital form is useful for operations such as applying the rotor brake.

Synthetic speech output was used to reinforce the colour changes, and was triggered as soon as the speed went out of the first set of limits.

The form of the torque readout was a vertical strip, with a digital readout below it. Normally the strip was white, and it changed to magenta when it exceeded the current torque limit (either 3000 or 2700 lb/ft depending upon number of engines running). The colour change was again backed up with an audio warning, which was repeated every seven seconds as long as the limit was exceeded.

The primary flight display format was found to be very effective. All information which was related to the collective - torque, height, rate of climb/descent and the height director - was displayed on the left-hand side of the display. At all times the pilot could see how much torque he was using, and how much he had available.

The inclusion of any more engine data on the PFD was not considered necessary, and would only have cluttered an already busy display.

8 CONCLUSIONS

These trials have shown that there is considerable benefit in not presenting engine data to the pilot at all times. Suppression of engine data, except when required, enables one less display surface to be installed in cockpits which will be crowded anyway. If a

dedicated engine display were installed, pilots would continue to scan the display and monitor the engines, a task far more suited to the microprocessors that are readily available to the systems designer.

Data on the engine display should be grouped according to engine, and not parameter type. Strip gauges give a good packing density whilst maintaining legibility. Bold colour changes, nonlinear scaling and 'tram lines', to indicate parameter operating margins, all aid in rapid fault detection.

The use of synthetic voice output to give warnings has shown that headout time can be significantly increased when compared to the use of simpler audio warnings such as bells or klaxons. For warnings requiring a simple response such as a torque exceedance, the pilot does not need to bring his head back into the cockpit - for more complex problems the voice directs the pilot to the problem area, aiding rapid fault correction.

The only engine related data that the pilot needs constant access to are torque and rotor rpm. These need to be incorporated on the primary flight display. Here again, strip gauges have been found to be effective.

The ability to calculate power performance indices in flight in a simple form has been demonstrated and found valuable. The use of a better engine model should enable true real time monitoring to be performed, with potential savings through both increased engine life and reduced servicing time.

The engine monitoring system is just one part of an integrated cockpit display and monitoring system which has been shown to offer significant benefits over conventional helicopter avionics. The system shows great potential for increasing mission effectiveness and aircrew safety, and reducing aircraft servicing times.

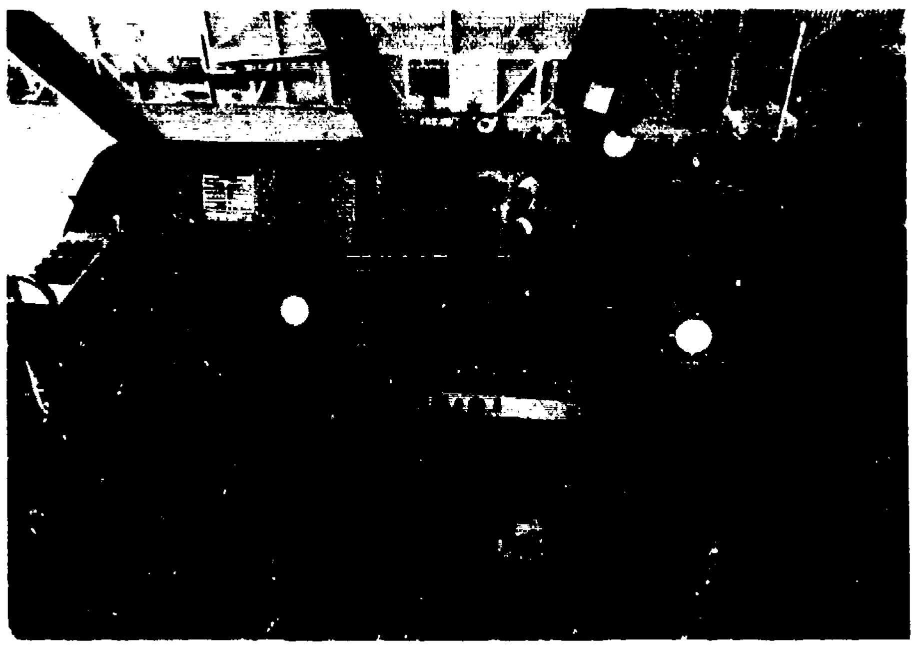

FIGURE 1 THE WESSEX COCKPIT.

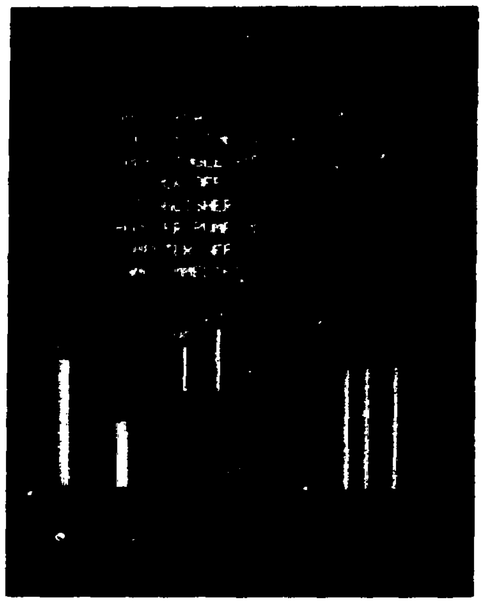

FIGURE 2 EARLY ENGINE FORMAT.

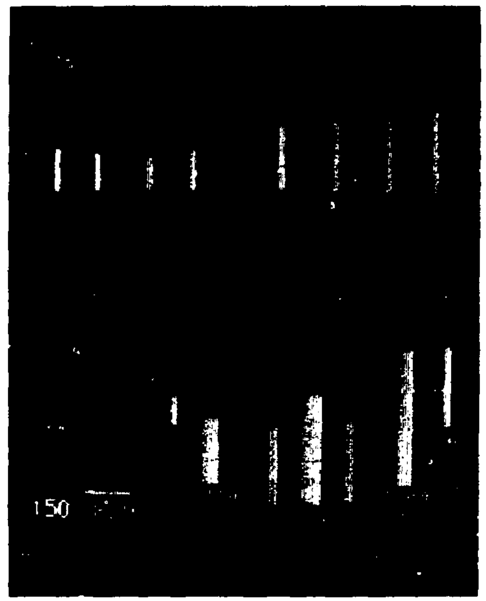

FIGURE 3 CURRENT ENGINE FORMAT.

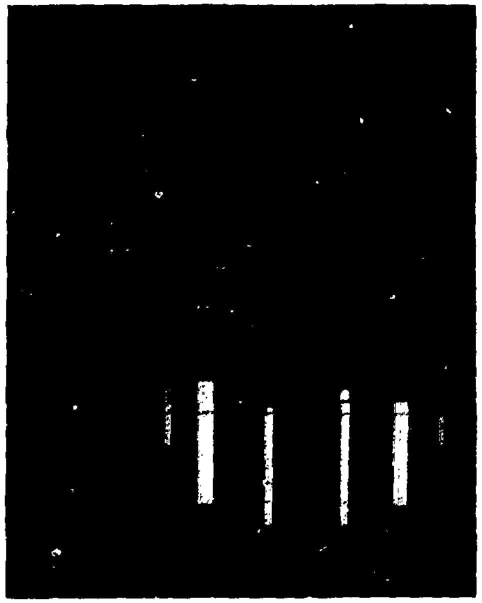

FIGURE4. TAKE OFF CHECKS.

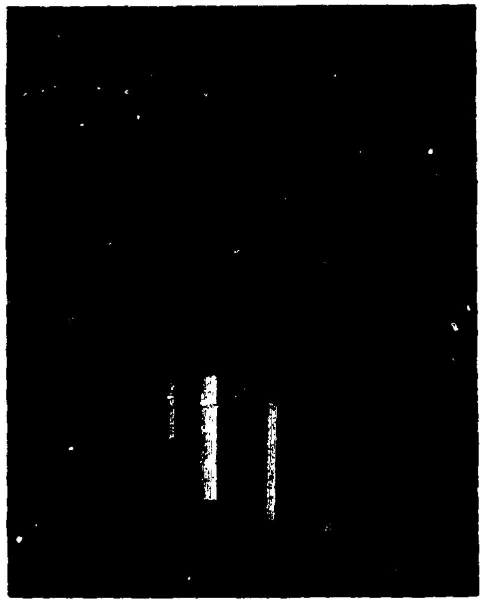

FIGURE 5. ENGINE FAILURE.

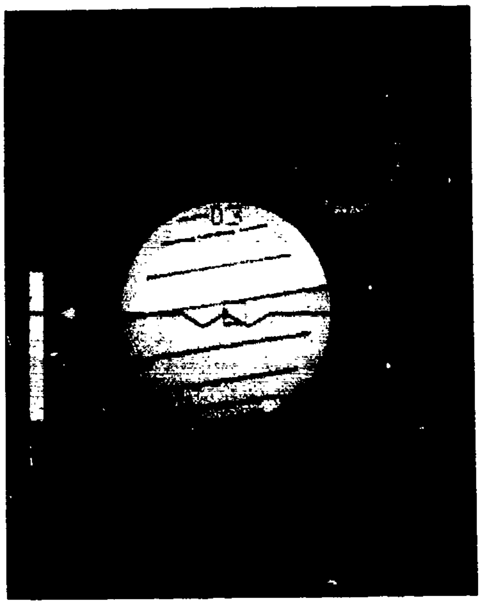

FIGURE 6. PRIMARY FLIGHT DISPLAY

CONTROLE ACTIF DES VIBRATIONS SUR HELICOPTERE PAR COMMANDES MULTICYCLIQUES AUTOADAPTATIVES

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RESUME

Un programme de Recherche, soutenu par les Services Officiels Français, a été mené à l'AEROSPATIALE, Division Hélicoptères, avec pour objectif le développement d'un système probatoire de contrôle actif des vibrations par commandes multicycliques appliquées aux pales du rotor principal.

Cette expérimentation constitue une des applications les plus significatives des techniques de commande autoadaptative des systèmes stochastiques.

Les différentes étapes de ce programme sont exposées, depuis la recherche d'une modélisation théorique du comportement vibratoire de l'hélicoptère, sous l'effet de commandes multicycliques, jusqu'aux essais en vol du système.

La campagne d'essais, qui s'est déroulée en 1985 à l'AEROSPATIALE Marignane sur GAZELLE SA 349, a permis ainsi de valider le concept de réduction des vibrations par un système autoadaptatif en boucle fermée, dans l'ensemble du domaine de vol de l'hélicoptère SA 349.

INTRODUCTION

De nombreux processus industriels sont commandés par des algorithmes de type P.I.D. (Proportionnel, Integral, Dérivé). Leur simplicité, leur robustesse, leurs bonnes performances dans de nombreux cas, expliquent leur succès. Le choix souvent aisé des paramètres de ces régulateurs s'appuie sur une connaissance grossière du processus lui-même.

Dans de nombreux domaines, ces méthodes classiques montrent rapidement leurs limites. Aucun régulateur à paramètres constants ne peut, en effet, prendre en charge les évolutions temporelles lentes ou rapides du système à commander.

Parallèlement à ces exigences, les développements récents de la mini et de la micro-informatique rendent possible l'implantation de lois de commandes complexes qui demandent un traitement substantiel.

Ces développements technologiques, associés aux exigences de certains cahiers des charges, justifient l'intérêt actuel des commandes autoadaptatives. Par autoadaptation, on entend une «régulation à paramètres variables», qui assure la minimisation d'un critère de performance pour des systèmes évolutifs.

La réduction des vibrations sur hélicoptère par commandes multicycliques est une application typique de régulateur autoadaptatif des systèmes stochastiques, menée depuis les études théoriques en simulation, jusqu'à la validation en vol.

De plus, le développement en parallèle d'un algorithme de type déterministe a été réalisé dans le but de quantifier le rapport performance/complexité pour deux approches différentes du problème.

LES VIBRATIONS SUR HELICOPTERE

Sur hélicoptère, les problèmes posés par les vibrations engendrées par les ensembles dynamiques sont importants, et lourds de conséquences (réduction de la durée de vie du matériel, contraintes de fiabilité, réduction du confort,.....).

Les moyens utilisés à ce jour pour limiter ces phénomènes sont des moyens passifs du type antivibreurs ou suspensions qui donnent des résultats acceptables dans beaucoup de cas. Toutefois, les exigences de plus en plus sévères en matière de confort, alliées à des objectifs de vitesses de croisière de plus en plus élevées, font que ces systèmes seront limités dans l'avenir, c'est-à-dire que le maintien du niveau vibratoire exigé risque d'entraîner des augmentations rédhibitoires de leur masse.

Parallèlement aux moyens passifs, des moyens actifs de contrôle des vibrations, dont la commande multicyclique est un cas particulier, sont envisagés (Réf. : (1) à (13)).

Le contrôle multicyclique permet, par une action directe au niveau de la commande de pas des pales, de minimiser les vibrations induites dans la structure à une fréquence caractéristique.

En effet, sur un hélicoptère à rotor tripale, la fréquence prépondérante des vibrations dans la cellule est 3Ω (Ω fréquence de rotation du rotor). Ces vibrations ont pour origine des efforts alternés à la fréquence 3Ω selon l'axe du rotor, transmis directement à la cellule, et des efforts aux fréquences 2Ω et 4Ω dans le plan du rotor, transmis à la cellule, après changement de repère, en efforts de fréquence 3Ω (Figure 1a).

Des commandes, générées en série avec les ordres de pilotage, à la fréquence 3Ω, créent des efforts en 2Ω, 3Ω et 4Ω au niveau du rotor, qui peuvent s'opposer à ceux qui occasionnent des vibrations (Figure 1b).

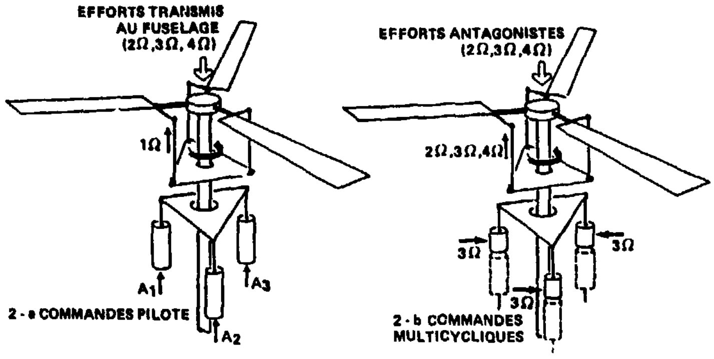

Fig. 1 : PRINCIPE DU CONTROLE MULTICYCLIQUE EN REPERE FIXE

Ainsi, le système multicyclique a pour tâche d'identifier le transfert entre commandes multicycliques et vibrations (variable en fonction du cas de vol et de la configuration appareil), de façon à calculer le module et la phase de chacune des trois commandes optimales à appliquer aux vérins multicycliques, afin de réduire les vibrations dans la cellule.

Ceci conduit au schéma fonctionnel présenté Figure 2. L'analyse harmonique permet d'extraire les coefficients de Fourier correspondant à la fréquence prépondérante (c'est-à-dire 3Ω) des mesures vibratoires. A partir de ces informations et de la connaissance des commandes multicycliques précédentes, le calculateur numérique calcule les modules et phases des trois commandes multicycliques. Celles-ci sont transformées par le synthétiseur en trois signaux sinusoïdaux à la fréquence 3Ω, commandant les vérins multicycliques.

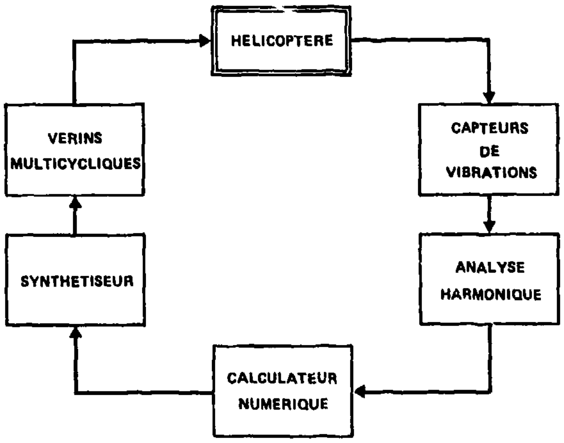

Fig. 2 : SCHEMA FONCTIONNEL DU SYSTEME MULTICYCLIQUE

La validité théorique du concept, et une première quantification des gains potentiels, ont été obtenues par simulation numérique du rotor associée à des essais sur la réponse structurale de la cellule. Elles ont été corroborées par des essais simplifiés sur banc rotor en 1977 à l'AEROSPATIALE.

Un programme de recherche, partiellement soutenu par les Services Officiels Français, a été lancé en 1980 par la Division Hélicoptères de l'AEROSPATIALE avec pour objectif l'expérimentation d'un système probatoire de contrôle actif des vibrations par commandes multicycliques sur l'appareil de recherche tripale SA 349, dérivé de la GAZELLE SA 342 (Figure 3).

Fig. 3 : HELICOPTERE EXPERIMENTAL SA 349

LE SYSTEME EXPERIMENTAL

Compte tenu des exigences en matière de performances et de sécurité, un système du type «simplex surveillé» a été retenu (Figure 4), la commande multicyclique étant limitée en amplitude à+/−1,7 degré.

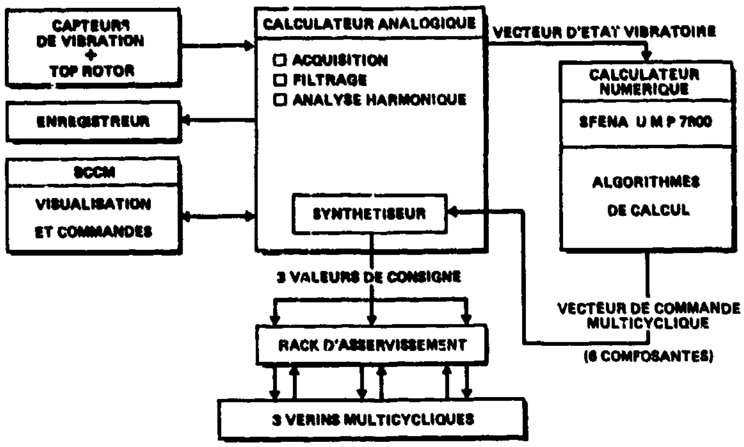

Fig. 4 : ARCHITECTURE DU SYSTEME EXPERIMENTAL

Le système expérimental comprend principalement :

- des capteurs de vibrations (accéléromètres placés en différents points de la cabine) et un capteur magnétique du top rotor (pour la connaissance précise de Ω, et la synchronisation) ;
- un calculateur analogique réalisant l'analyse harmonique (extraction de la composante des vibrations en 3Ω), la génération des ordres en 3Ω (fonction synthétiseur) destinés au rack d'asservissement des vérins multicycliques, et la gestion des sécurités ;
- un calculateur numérique dans lequel des algorithmes de calcul élaborent le «vecteur de commande» optimal à partir du «vecteur vibratoire» provenant du calculateur analogique ;
- un rack d'asservissement des vérins multicycliques ;
- trois vérins électrohydrauliques (dits vérins «multicycliques») montés en série sur les servo-commandes classiques à entrée mécanique ; leur course est limitée à 10 mm, correspondant à un pas de pale de+/−1,7 degré. Ces vérins ont été développés pour cette application, afin d'obtenir de bonnes performances à des fréquences de commande élevées (3Ω soit 19 Hz pour l'hélicoptère SA 349) et sous des charges dynamiques importantes ;
- un boîtier de commande placé dans la cabine, interface entre le système et l'équipage d'essai.

Les différents éléments constituant le système multicyclique ont été développés selon les spécifications de l'AEROSPATIALE, par les sociétés françaises GIRAVIONS DORAND (rack d'asservissement), AIR-EQUIPEMENT (vérins), SFENA (calculateur numérique) et AEROSPATIALE Division Hélicoptères pour les autres éléments.

La conception, la validation et la programmation du logiciel embarqué ont été réalisées par une équipe de la Direction des Etudes de l'AEROSPATIALE, avec la participation de l'ONERA (CERT/DERA) pour l'étude des algorithmes stochastiques.

MODELISATION DU PROBLEME

Trois algorithmes de calcul de la commande optimale ont été développés. Tous trois sont basés sur une représentation linéaire du transfert entre commandes multicycliques et vibrations cellule, résultat de la simplification de la modélisation du rotor et des essais sur la structure de l'appareil expérimental :

$$Z_k = S \cdot \theta_{k-1} + Z_o$$

avec :

- Z_o vecteur des 2n coefficients de Fourier en 3Ω correspondant à n mesures accélérométriques, sans commandes multicycliques.
- Z_k vecteur de mesure au pas k, après commandes multicycliques.
- θ_{k-1} vecteur des 6 coefficients de Fourier en 3Ω correspondant aux commandes appliquées aux 3 vérins au pas k - 1.
- S matrice représentative de la sensibilité du vecteur vibratoire au vecteur de commande multicyclique (dimension : 2n lignes, 6 colonnes).

Le vecteur de commande θ_k est calculé à chaque pas par minimisation d'un critère quadratique J :

$$J \quad Z^T{}_{k+1} \cdot Z_{k+1} + \Delta\theta^T{}_k \cdot W \cdot \Delta\theta_k$$

faisant intervenir, d'une part l'énergie des vibrations à diminuer ($Z^T{}_{k+1} \cdot Z_{k+1}$), d'autre part un terme de pondération sur la variation de commande ($\Delta\theta^T{}_k \cdot W \cdot \Delta\theta_k$ avec W matrice définie, positive) permettant une action progressive et donc «prudente» sur le système.

L'algorithme a donc pour tâches :

– d'identifier la matrice S à chaque instant puisqu'elle dépend des conditions de vol et de configuration de l'appareil ; l'identification de Z_o n'est pas indispensable dans la mesure où la commande optimale est calculée de façon itérative en utilisant un modèle local :

$$\Delta Z_k = S \cdot \Delta\theta_{k-1}$$

– de calculer la variation de commande optimale $\Delta\theta_k{}^*$.

ETUDE DES ALGORITHMES

Trois algorithmes de deux types différents ont été étudiés :

– l'Algorithme Adaptatif Déterministe (AAD).
– le Régulateur Adaptatif Stochastique (RAS).
– le Régulateur Adaptatif Stochastique avec Estimation des Vibrations (RASEV).

L'algorithme AAD est de type déterministe. Il réalise l'identification de la matrice S par l'envoi de commandes «calibrées» ou extra-signaux. Après la première identification lors de l'initialisation, la comparaison entre le niveau vibratoire mesuré et un niveau vibratoire estimé (par calculs) permet de déterminer s'il est nécessaire d'identifier à nouveau S.

Cet algorithme implique donc :

– une excitation importante du système lors des phases d'identification,
– le choix d'un critère pour identifier S uniquement lors d'une modification des conditions de vol. En effet, il est nécessaire de minimiser les phases d'identification, car elles nécessitent l'envoi de 6 commandes calibrées, a priori non optimales dans le sens de la diminution des vibrations.

En dehors des phases d'identification, le calcul de la commande optimale est réalisé à chaque pas par minimisation du critère J, en considérant que S a bien été identifiée :

$$\partial J / \partial(\Delta\theta_k) = 0, \text{ d'où :}$$

$$\Delta\theta^*{}_k = -(W + S^T \cdot S)^{-1} \cdot S^T \cdot Z_k$$

L'algorithme RAS, de type stochastique, utilise la connaissance «a priori» des caractéristiques statiques des bruits de mesure et de système pour identifier S à chaque pas. Il est constitué de 2n filtres de Kalman, chacun identifiant une ligne de la matrice S.

Pour l'équation d'état des filtres, l'hypothèse retenue est la faible variation de S entre deux pas successifs, qui se traduit par la constance de la matrice S, au bruit d'état près.

L'équation de mesure découle de la modélisation en variations :

$$\Delta Z_k = S \cdot \Delta\theta_{k-1}$$

Les caractéristiques des bruits de mesure et d'état étant supposées ne pas dépendre de la ligne de la matrice S, les 2n filtres de Kalman possèdent les mêmes matrices de covariance d'erreur P_k, et gains de Kalman K_k. Seule l'équation d'évolution dépend de la ligne identifiée.

Pour tenir compte d'une faible évolution de S entre deux pas successifs, un paramètre d'oubli a été introduit dans l'équation de recalage de la matrice de covariance de l'erreur.

L'initialisation de l'algorithme est effectuée par l'envoi de commandes aléatoires de faible amplitude.

L'identification de la matrice S étant réalisée à chaque pas en utilisant la variation de commande précédente, cette dernière peut être calculée pour être optimale.

La connaissance «a priori» des caractéristiques statistiques de la matrice S identifiée, permet, pour le calcul de la variation de commande optimale $\Delta\theta_k$, de choisir le critère stochastique suivant :

$$J = E\,(Z^T{}_{k+1} \cdot Z_{k+1} + \Delta\theta^T{}_k \cdot W \cdot \Delta\theta_k)$$

E (.) : Espérance mathématique

Ce qui conduit à :

$$\Delta\theta^*{}_k = -(W + S^T \cdot S + 2n \cdot P_k)^{-1} \cdot S^T \cdot Z_k$$

Par souci de simplicité, le principe d'équivalence a été retenu ($P_k = 0$), de façon à conserver la même commande optimale que dans le cas déterministe. Néanmoins, un caractère de prudence a été introduit par une loi de variation de W, croissante en fonction de la variation du niveau vibratoire.

Ainsi, l'algorithme RAS :

– permet une identification permanente de la matrice S, la commande multicyclique optimale étant envoyée à chaque pas,
– prend en compte les caractéristiques statistiques des bruits de mesure.

L'algorithme RASEV, est du même type que le précédent. Il ne diffère que par la prise en compte du modèle global :

$$Z_k = S \cdot \theta_{k-1} + Z_o$$

Il identifie donc S et Z_0 à chaque pas de calcul, à l'aide de 2n filtres de Kalman dont les vecteurs d'état sont constitués d'une ligne de la matrice S associée à la composante correspondante du vecteur Z_0.

Le calcul de la commande optimale est identique à celui réalisé par les deux autres algorithmes.

Le RASEV se caractérise donc par :

- une identification permanente avec envoi de commandes optimales à chaque pas,
- une meilleure prise en compte de la modélisation.

SIMULATIONS EN TEMPS DIFFERE

Une simulation linéaire du comportement vibratoire de l'hélicoptère sous l'effet de commandes multicycliques a permis la mise au point des algorithmes décrits précédemment. Cette simulation est caractérisée par un ensemble de cinq matrices S et vecteurs Z_0 (obtenus par calculs sur un modèle rotor, associés à des essais sur la cellule) correspondant à différents cas de vitesse longitudinale. La Figure 5 montre l'évolution d'une ligne de la matrice S pour une vitesse longitudinale variant de 200 km/h à 280 km/h, le «raccordement» entre deux vitesses successives étant réalisé au moyen de fonctions polynômiales.

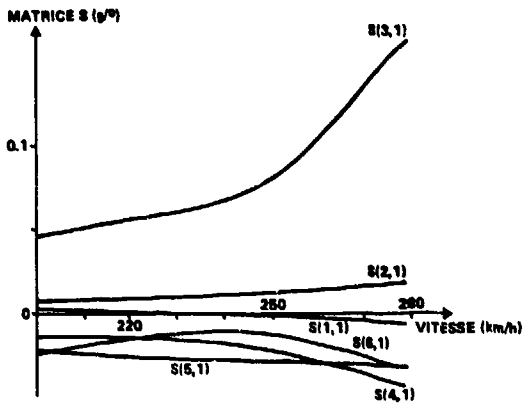

Fig. 5 : EVOLUTION D'UNE COLONNE DE LA MATRICE S UTILISEE EN SIMULATION

Un exemple des résultats obtenus par simulation des algorithmes en boucle fermée est présenté sur la Figure 6, montrant dans le cas d'une phase d'accélération (passage de 200 km/h à 280 km/h) l'influence de la commande multicyclique sur le niveau vibratoire moyen dans la cabine. Sur chacun des graphiques, est présentée l'évolution du niveau vibratoire, avec et sans commande multicyclique, pour l'hélicoptère muni de sa suspension passive.

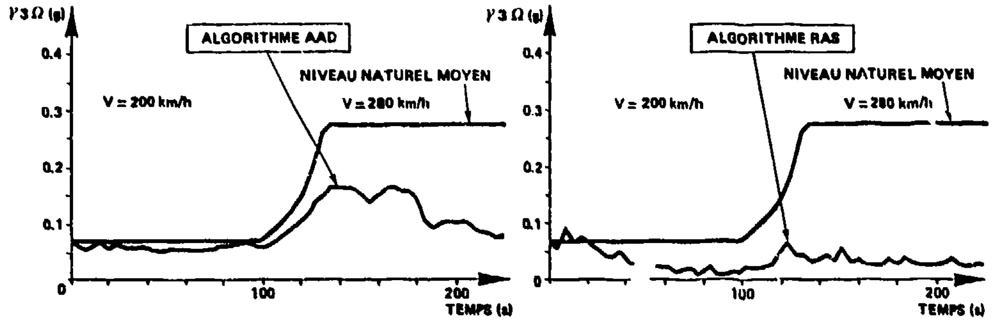

Fig. 6 : SIMULATION DES ALGORITHMES EN TEMPS DIFFERE

Dans la limite de la modélisation linéaire retenue, ces simulations ont permis de démontrer les bonnes performances d'autoadaptativité des algorithmes durant les phases d'évolution (surtout pour les algorithmes stochastiques), d'estimer les gains potentiels en vibrations, et d'évaluer l'influence des différents paramètres de réglage sur l'efficacité des algorithmes (rapidité de convergence, gains, autoadaptativité..).

ESSAIS EN VOL DU SYSTEME EXPERIMENTAL

Le système multicyclique a été évalué en vol pour deux configurations de la machine de base : suspension bidirectionnelle «libre» (correspondant à la GAZELLE SA 349 munie de son système antivibratoire passif) et suspension «bloquée» (correspondant à une machine sans filtrage passif des vibrations).

Pour chacune de ces configurations, les trois algorithmes multicycliques ont été testés en boucle fermée sur l'ensemble du domaine de vol. La course des commandes multicycliques a été limitée à+/–1 degré au cours de cette expérimentation, compte tenu des efforts dynamiques importants sur la chaîne de commande de vol, rencontrés au cours des essais d'identification. La course de+/–0,8 degré a été retenue

pour l'analyse comparative complète des trois algorithmes, une augmentation de course jusqu'à 1 degré ayant été effectuée pour l'algorithme RASEV uniquement.

La position des capteurs d'acquisition du système a fait l'objet d'une optimisation au cours de ces essais, conduisant à retenir quatre accéléromètres : deux, selon l'axe vertical et l'axe longitudinal, à l'avant de la cabine, deux, selon l'axe vertical, en places pilote et copilote.

Seront présentés ici les résultats obtenus avec le système actif agissant sur l'hélicoptère sans filtrage passif des vibrations (suspension «bloquée»), ce cas correspondant, très probablement, à l'utilisation que l'on peut envisager sur les hélicoptères futurs.

METHODOLOGIE DE CONDUITE DES ESSAIS

Les essais en vol du système expérimental ont été conduits selon une méthodologie basée essentiellement sur l'importance des simulations en temps différé, incluant un modèle représentatif du comportement vibratoire de l'hélicoptère sous l'effet de commandes multicycliques (Figure 7).

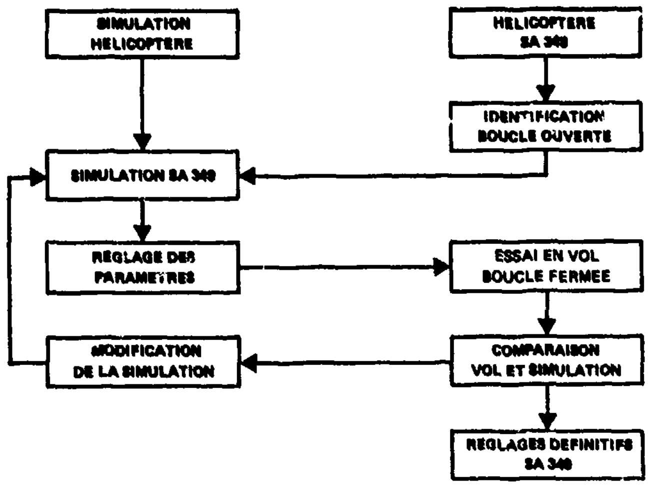

Fig. 7 : METHODOLOGIE DE CONDUITE DES ESSAIS

Aussi, le logiciel embarqué est-il constitué de différents modules, sélectionnables en vol à l'aide du boîtier de commande, permettant l'identification complète de l'hélicoptère SA 349 et les essais des trois algorithmes :

- Mesures sans commandes multicycliques
- Séquences d'échelons de commande calibrés (5 niveaux possibles)
- Algorithme AAD
- Algorithme RAS — Deux jeux de paramètres possibles
- Algorithme RASEV

IDENTIFICATION EN BOUCLE OUVERTE

La phase d'identification, étape prépondérante dans cette méthodologie, a été conduite au cours de vols spécifiques grâce aux deux premiers modules du logiciel embarqué : mesures sans commandes multicycliques et séquences d'échelons de commande calibrés.

Elle a permis de constituer une «banque de données» importante de l'effet des commandes multicycliques, utile pour la mise au point en simulation des algorithmes. La Figure 8 est un exemple de courbes, obtenues en vol stabilisé à 160 km/h, représentant l'évolution des composantes du vecteur vibratoire Z en fonction de l'amplitude d'une des composantes du vecteur de commande θ.

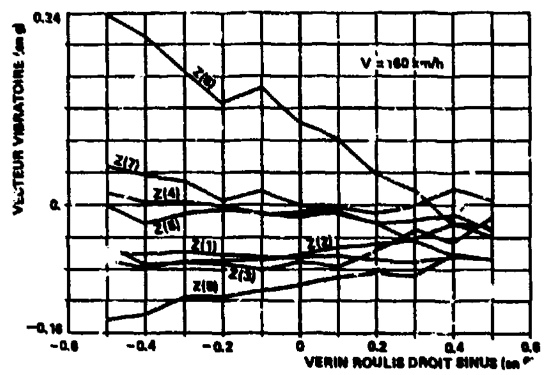

Fig. 8 : IDENTIFICATION EN VOL DU COMPORTEMENT VIBRATOIRE DE L'HELICOPTERE

REGLAGE DES PARAMETRES EN SIMULATION

Les paramètres de réglage des algorithmes multicycliques ont été obtenus après simulations en temps différé. La similitude entre l'hélicoptère et sa représentation en simulation (vis-à-vis du comportement vibratoire) a permis de reconduire ces réglages lors des essais en vol.

La Figure 9 correspond à une comparaison entre la simulation (trait plein) et le vol (trait en pointillé) pour une composante du vecteur de mesure, lors d'un essai constitué de paliers successifs à différentes vitesses.

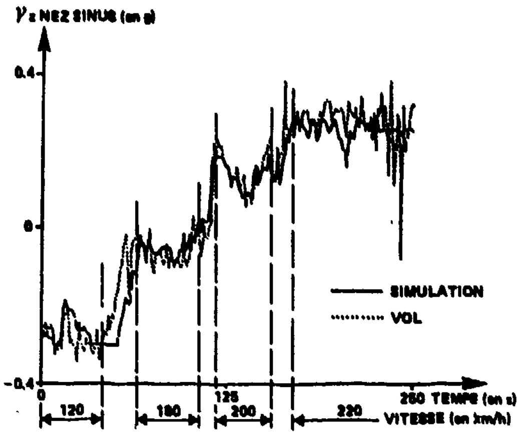

Fig. 9 : COMPARAISON VOL/SIMULATION A ISO-CONDITIONS (PALIERS A DIFFERENTES VITESSES)

Cette méthodologie a ainsi permis :

- d'accéder rapidement aux essais en vol du système en boucle fermée ; ainsi, un délai de seulement trois semaines après le vol d'identification aura été nécessaire pour débuter les essais en boucle fermée.
- de minimiser le nombre de vols grâce aux réglages préliminaires obtenus en simulation.

ESSAIS DES ALGORITHMES MULTICYCLIQUES EN VOL (BOUCLE FERMEE)

La procédure d'essai adoptée pour la mise au point et la comparaison des trois algorithmes, a consisté en une succession de paliers stabilisés à différentes vitesses, le système restant actif lors des phases d'accélération entre les paliers. Cette procédure a ainsi permis de tester les performances des algorithmes, à la fois pour la réduction des vibrations, et pour le critère d'autoadaptativité (prise en compte d'évolution rapide du cas de vol).

Après mise au point, les algorithmes ont été évalués dans tout le domaine de vol de la GAZELLE SA 349.

La comparaison des trois algorithmes, présentée en Figure 10, a été obtenue au cours d'un vol en boucle fermée, avec la procédure d'essai décrite précédemment (le Niveau Vibratoire Global correspond à la moyenne quadratique de la composante en 3Ω des mesures effectuées sur les capteurs utilisés par le système).

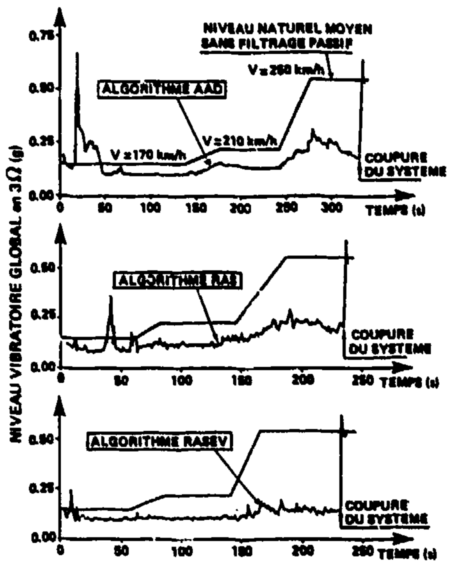

Fig. 10 : RESULTATS OBTENUS EN VOL PAR LES ALGORITHMES MULTICYCLIQUES (COURSE DE COMMANDE MAXIMALE : 0,8 DEGRE)

On voit ainsi que les gains en vibration, obtenus avec les trois algorithmes, sont assez proches (de l'ordre de 80 % à 250 km/h), l'algorithme RASEV étant le plus efficace. La Figure 11 détaille les niveaux vibratoires obtenus à 250 km/h avec les trois algorithmes testés, et également sans aucun système de filtrage des vibrations (hélicoptère de base), mesurés selon l'axe vertical en trois points de la cellule.

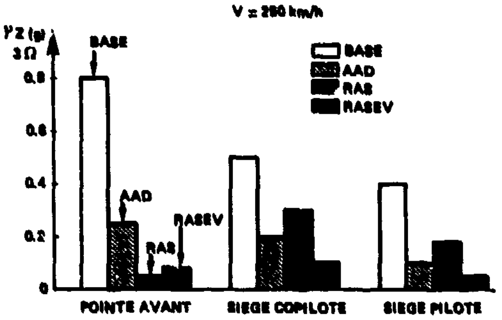

Fig. 11 : COMPARAISON DES TROIS ALGORITHMES TESTES (COURSE DE COMMANDE MAXIMALE : 0,8 DEGRE)

Une partie des différences constatées, entre les algorithmes stochastiques (RAS et RASEV) et l'algorithme déterministe (AAD), s'explique par les différences de course «efficace» (course utilisable pour la commande optimale). Ainsi pour une même course maximale, l'algorithme AAD a une course efficace réduite (réduction de l'ordre de 0,2 degré) afin de conserver une marge pour les échelons d'identification.

En effet, au cours de ces essais, il a été démontré que les gains en vibrations étaient directement reliés à la course autorisée pour la commande optimale.

L'influence de la course de commande sur les vibrations en cabine en 3Ω est présentée sur la Figure 12, pour les trois algorithmes et trois vitesses de palier. En extrapolant les courbes, on peut en déduire que des gains en vibrations plus importants pourraient être obtenus avec le système multicyclique avec des courses de commandes plus importantes.

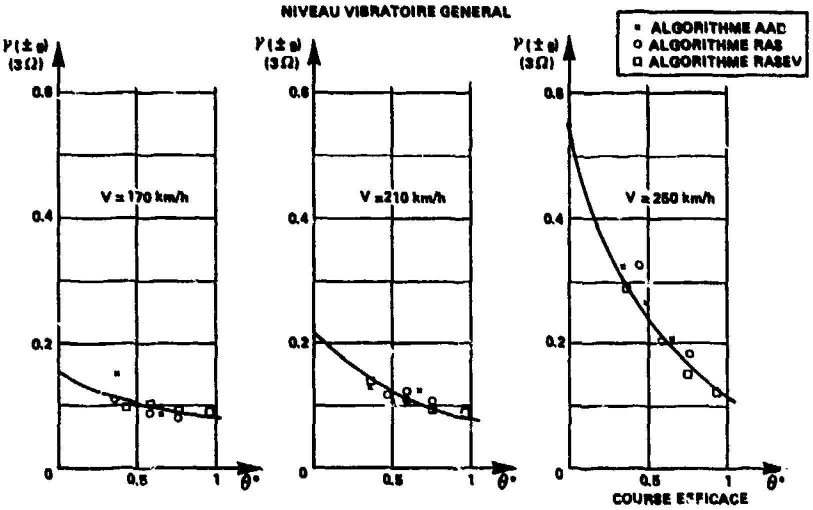

Fig. 12 : INFLUENCE DE LA COURSE DE COMMANDE (COURSE DE COMMANDE MAXIMALE : 1 DEGRE)

Mais, rappelons-le, la réduction des vibrations n'est pas le seul critère de choix des algorithmes.

Les performances d'autoadaptativité ont aussi de l'importance pour le choix définitif d'un algorithme puisqu'elles influent directement sur le confort des passagers, ceux-ci étant particulièrement sensibles aux variations brutales du niveau vibratoire.

Vis-à-vis de ce critère, l'algorithme déterministe AAD présente quelques faiblesses : l'identification engendre des «pics» en vibrations élevés lors de la mise en route de l'algorithme (Figure 10), jusqu'à identification satisfaisante de la matrice S.

Néanmoins, après optimisation des paramètres, les séquences d'identification ne se déclenchent que lors de modifications du cas de vol (accélérations), et n'engendrent pas nécessairement des pics importants en vibrations, le sens de variation de chaque commande étant choisi en fonction de la matrice S précédente, afin d'aller dans le sens de la diminution du Niveau Vibratoire Global.

Les algorithmes stochastiques, compte tenu de l'identification permanente de la matrice S (et de Z_0 pour le RASEV), ont montré de très bonnes performances d'autoadaptativité.

L'exemple caractéristique présenté Figures 13 et 14, correspond à une mise en virage (facteur de charge n_z : 1,5 g) à la vitesse de 200 km/h, l'algorithme RASEV étant en fonctionnement avec une autorité de commande multicyclique de +/−0,8 degré.

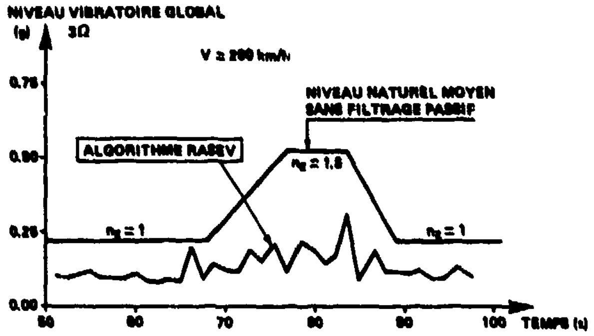

Fig. 13 : REPONSE DU SYSTEME LORS D'UNE MISE EN FACTEUR DE CHARGE (ALGORITHME RASEV AVEC COURSE DE COMMANDE MAXIMALE : 0,8 DEGRE)

La Figure 13 montre que le niveau vibratoire n'a pas été perturbé lors de la mise en virage de l'hélicoptère, la Figure 14 permettant de démontrer que cette stabilité a été obtenue grâce à la modification de la matrice S au cours du virage (est présentée la sensibilité d'une composante du vecteur vibratoire à la variation de la commande multicyclique en tangage et l'évolution de la composante correspondante du vecteur de commande).

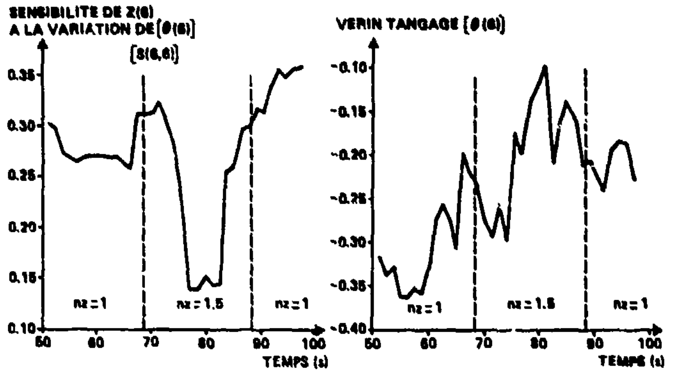

Fig. 14 : ETUDE DE L'AUTOADAPTATIVITE LORS D'UNE MISE EN FACTEUR DE CHARGE (ALGORITHME RASEV AVEC COURSE DE COMMANDE MAXIMALE : 0,8 DEGRE)

COMPARAISON AVEC UNE SUSPENSION PASSIVE

Si l'on compare les performances du système actif de contrôle des vibrations avec celles de la suspension passive de la GAZELLE SA 349, (Figure 15), on constate que le système actif conduit à des niveaux de vibrations équivalents au système passif là où ce dernier est le plus efficace (sièges pilote et copilote notamment), le système actif étant nettement plus performant aux autres postes (pointes avant et arrière cabine).

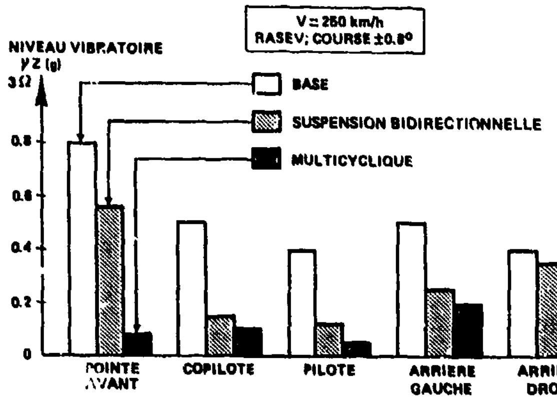

Fig. 15 : COMPARAISON AVEC UN SYSTEME ANTIVIBRATOIRE PASSIF

Sur la même figure, on vérifie que le système multicyclique agit non seulement aux emplacements correspondant aux mesures intervenant dans son optimisation, mais aussi en des points non pris en compte par les algorithmes (arrière cabine), ceci étant dû à l'action du système directement à la source des vibrations (efforts tête rotor).

CONCLUSION

Le développement du système expérimental de contrôle actif des vibrations par commandes multicycliques a conduit à prouver l'efficacité d'un système en boucle fermée pour la réduction des vibrations sur l'ensemble du domaine de vol d'un hélicoptère.

La comparaison des résultats obtenus pour les deux types d'algorithmes testés, tend à montrer l'efficacité des algorithmes stochastiques. Leur complexité, largement compensée par une meilleure prise en compte des évolutions des paramètres de vol, n'est pas rédhibitoire pour les calculateurs numériques embarquables.

Les suites de cette action, envisagées dès ce jour, concernent notamment des études d'avant-projet de systèmes «série», afin d'évaluer les coûts d'un tel système pour un hélicoptère de série.

Enfin, cette expérimentation constitue, outre une des premières réalisations concernant la théorie de la commande optimale autoadaptative, une application importante des techniques numériques sur hélicoptère, et devra déboucher sur d'autres aspects du Contrôle Automatique Généralisé sur Hélicoptère (CAGH).

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CONTROLLING THE DYNAMIC ENVIRONMENT DURING NOE FLIGHT

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SUMMARY

A six degree-of-freedom (DOF) isolation system using six LIVE units has been installed under an Army/NASA contract on a Bell 206LM helicopter. This system has been named the Total Rotor Isolation System, or TRIS. To determine the effectiveness of TRIS in reducing helicopter vibrations, a flight verification study was conducted at Bell's Flight Research Center in Arlington, Texas. The objective was to demonstrate a 90% (or greater) isolation of the helicopter fuselage from the main rotor forces and moments transmitted at the blade passage frequency, which is 4/rev in the case of the Bell 206LM. The flight test was the final phase of a program performed by Bell under a NASA Langley Research Center contract with funding by the U.S. Army Aerostructures Directorate. The flight test data from the testbed aircraft indicate that the program objectives have been surpassed; the 4/rev vibration levels at the pilot's seat were suppressed below the 0.04g level throughout the transition envelope, which traditionally has high vibration levels. Results of flight tests to date indicate over 95% suppression of vibration levels from the rotor hub to the pilot's seat at a considerable weight savings over traditional antiresonant isolation concepts. In addition, the TRIS installation was designed with a decoupled control system and has shown a significant improvement in aircraft flying qualities. The improvement was such that it permitted the trimmed aircraft to be flown "hands-off" for a significant period of time, over 90 seconds. In conclusion, the TRIS flight test program has demonstrated a system that meets the objective of greatly reducing vibration levels of a current-generation helicopter, the Bell 206LM, while significantly improving the flying qualities to a point where stability augmentation is no longer a requirement.

INTRODUCTION

Vibratory excitations, inherent to the helicopter, cause many undesirable effects. These include helicopter crew fatigue, resulting in decreased proficiency; unacceptable passenger comfort; poor component and system equipment lives; lower avionics reliability, resulting in increased operating costs; and, in many cases, limited operational envelopes.

Vibration reduction has been a major goal of the rotary wing community since the helicopter's inception. In the 1970s, in recognition of the adverse effects of vibration and desiring a more stable weapons platform, the military reduced the MIL-SPEC acceptable levels of the predominant rotor harmonic (n/rev) g-levels from 0.15g at cruise speed to 0.05g. Commercial operators, particularly those conducting long flights to offshore oil rigs or ambulance runs, have also demanded lower vibration levels in aircraft.

Paralleling the desire for lower vibration levels, new objectives for high-speed performance, higher payloads, improved maneuverability, and increased agility have also been demanded by the helicopter users. These new goals have led to new rotor designs, including soft-in-plane, rigid, articulated with large hinge offsets, and teetering rotors with added hub springs. All of these changes tend toward increased weight and the generation of higher excitation shears and/or moments by the rotor.

Helicopters using first-generation main rotor shaft isolation systems of the 1940s and 1950s exceeded 0.5g n/rev vibration levels. Second-generation designs of the 1960s, with focal pylons, were generally able to meet the new 0.15g vibration level requirement at cruise, but not during transition. During the 1970s, several antiresonant isolation concepts were developed to isolate the fuselage from the primary source of vibration - the main rotor oscillatory forces. These devices include Kaman's DAVI, Boeing-Vertol's IRIS, and Bell's Nodal Beam. All of these concepts use a spring and a mechanically amplified mass to develop isolation or effect force cancellation at the main rotor excitation frequency. These systems were designed to meet the 0.05g level, but failed. The weight penalties caused by these systems or a combination of these systems varied from 2% to 3% or more of the helicopter design gross weight. It should be noted that the current state-of-the-art Army helicopters, the Sikorsky UH-60 Blackhawk and the McDonnell Douglas AH-64 Apache, never met the vibration criterion during competition.

With the overall objective of meeting the Army's MIL-SPEC vibration objective and reducing the helicopter's overall weight, the U.S. Army's Aerostructures Directorate (then the Army's Structures Laboratory), located at NASA's Langley Research Center, issued a request for proposal in 1979 for the "Analysis of the Feasibility of a Six Degree-of-Freedom (6-D.O.F.) Isolation System." Under a NASA/Army contract, Bell completed an analysis[1] and was subsequently awarded a follow-on contract for the "Design, Analysis, Fabrication, and Bench Testing of a Total Main Rotor Isolation System (TRIS)." The results of the TRIS bench test were so promising that in 1984 the decision was made to install the system on Bell's Model 206LM helicopter and then ground- and flight-test the aircraft. This paper will report on the results of the ground and flight tests. The reader is referred to the NASA contractor report[2,3] for more details on the test program.

LIVE Isolation

In 1972, research was begun at Bell on the use of a hydraulic fluid in cylinders with different areas to amplify the motion of a tungsten piston that acted as a tuning weight. This concept progressed to a very compact system that used a high-density, low-viscosity liquid (mercury) as both the "hydraulic fluid" and the tuning weight.

The Liquid Inertia Vibration Eliminator (LIVE) unit is shown schematically in cross section in Fig. 1. An inner cylinder is bonded to an outer cylinder with a layer of rubber, as in a coaxial bushing rubber spring. Cavities at the top

and bottom are enclosed, creating reservoirs for the "hydraulic fluid." The inner cylinder is attached to the transmission, and the outer cylinder is attached to the fuselage. A hole or "tuning port" through the inner cylinder connects the upper and lower reservoirs.

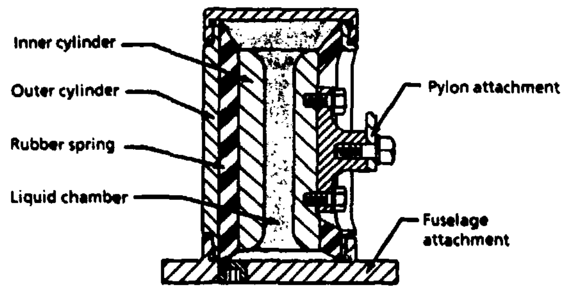

Fig. 1. LIVE system internal design.

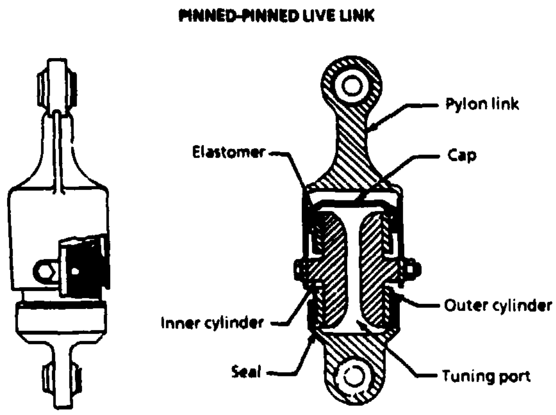

Fig. 2. Cutaway view of LIVE link.

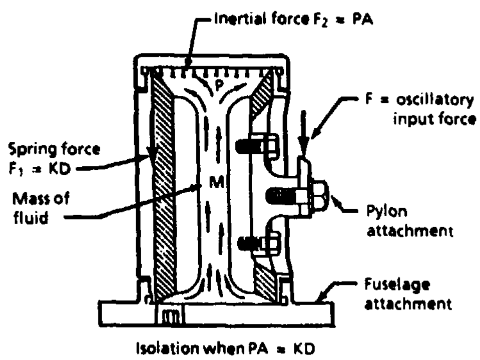

Fig. 3. Isolation principle of LIVE.

In operation, the liquid mercury oscillates within the LIVE units, and isolation is achieved when the pressure created by the inertial force due to motion of the mercury cancels the spring force due to the displacement of the rubber. By altering the spring rate and port diameter, the LIVE units can be tuned to isolate the desired blade passage frequency.

The mechanics of a classical pinned-pinned link is such that only axial loads can be transmitted; no moments can be input through the spherical bearings at its ends. If a LIVE unit is mounted within a link and tuned to completely isolate the blade passage frequency, then no oscillatory loads at the blade passage frequency (n/rev), in any direction, will be transmitted through the link. By using six pinned-pinned isolator links or LIVE units for attaching the pylon to the fuselage (in any configuration that is statically stable in all six degrees of freedom) with no other attachments, then every attachment will isolate the blade passage frequency and no oscillatory loads will be transmitted from any degree of freedom.

A representative LIVE isolator for the six degree-of-freedom application is shown in the cross-section view of Fig. 2. The inner member is attached to the pylon, and the outer member is attached to the fuselage. The two members are bonded to the elastomer that fills the annulus between them. This elastomer (working in shear) acts as a spring that supports and reacts to the static and dynamic loads placed on the isolator. Pressurized liquid mercury fills the center port in the inner member and both cavities at the ends of the isolator. No air space remains in the isolator. The action of the LIVE unit is illustrated in Fig. 3.

Baseline Helicopter

The helicopter selected for the purpose of establishing specific isolation system performance, weight analysis, system integration studies, and flight tests at minimum risk was the Bell 206LM helicopter, S/N 45269. A photo of the testbed aircraft is shown in Fig. 4. This is a 4000-lb class, turbine engine helicopter with a four-bladed, soft-in-plane, flexbeam rotor system. The isolation system as installed on the helicopter can be seen in Fig. 5.

GROUND VIBRATION TEST

The method selected to verify the performance of the TRIS was a ground vibration test (GVT), performed at Bell's Flight Research Center in Arlington, Texas. The GVT was designed to measure the isolation of each degree of freedom independently and, thus, without the influence of non-rotor-hub n/rev sources, as would be the case in flight.

Three different systems were used for hub excitation during the GVT. A single 1500-lb capacity electromagnetic shaker was used for hub vertical, lateral, and longitudinal shear inputs; two 1500-lb electromagnetic shakers were operated out of phase for hub yaw moment input; and a rotary hydraulic shaker was used for hub pitch and roll moment inputs.

For each excitation degree of freedom, sweeps were made with the full hub weight to determine the placement of pylon and fuselage natural frequencies and the approximate shape and frequency placement of the isolation valley. In addition, frequency dwells at various load levels with and without the hub weight at 1/rev, 4/rev, and 8/rev were made to determine the isolation efficiency and load linearity of the TRIS. Load levels up to 800 lb in shear, and 5000 in-lb in moment were applied to the hub.

Fig. 4. Bell Model 206LM Total Rotor Isolation System (TRIS) helicopter.

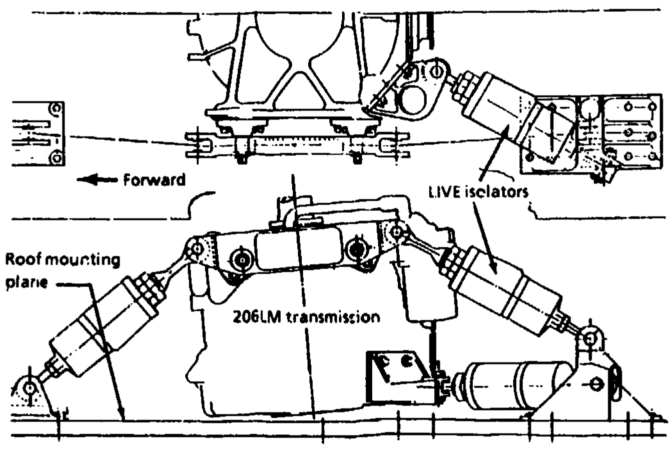

Fig. 5. Installation of six-degree-of-freedom isolation system on helicopter.

An array of six accelerometers at the hub and another array of six accelerometers near the fuselage cg were used to measure the input and response for each of the six degrees of freedom. These arrays were the primary transducers used to determine the percentage of isolation achieved by the TRIS and to determine if the system met the 90% isolation criterion of the contract statement of work. The accelerometer measurements were used to calculate the percentage of isolation in the six degrees of freedom in the following manner:

1. For the translation directions, the response of the two accelerometers, with their sensitive axis in the same direction, were averaged to determine the response of a point half way between them.

2. For the rotational directions, the response of the two accelerometers, with their sensitive axis in a plane perpendicular to the axis of rotation, were subtracted, one from the other, divided by the distance between them, and converted to units of deg/s^2. This calculation yielded the rotational response of the structure between the two accelerometers.

These calculations were performed by computer on both the sine and cosine components of the response so that correct phase and magnitude were maintained between the different transducers. In addition to the above accelerometers, triaxial accelerometers were located at each of the crew seats, each of the aft passenger seat locations, the elevator, and at the 90° gearbox.

Transfer functions were acquired on all accelerometers for each degree of freedom. These transfer functions were used with a Bell modal analysis computer program to define the natural frequencies of the pylon and the fuselage.

The major transfer functions for the pilot seat are presented in Figs. 6 through 11. These plots show that the TRIS valley at 4/rev (26.3 Hz) occurs in each accelerometer, and for each excitation degree of freedom. These figures show good frequency separation between 4/rev and all pylon and fuselage modes, although for this program no attempt was made to change the helicopter fuselage modes from the standard Bell Model 206L.

4/REV FORCED RESPONSE

For a more accurate measurement of the TRIS response at 4/rev, forced response data were acquired by exciting the aircraft with a constant 4/rev sine wave. This part of the test was performed with no hub weight so that the hub and airframe response would equal the in-flight response for the same hub load. By measuring hub response in g's or deg/s^2 and ratioing it to the cg response in the same units, a measure of the TRIS transmissibility was calculated.

The fuselage response data are plotted in Figs. 12 through 17. The dual scaling of these plots was selected to quickly show if the 90% isolation criterion was achieved. The hub response was plotted with the cg response for each degree of freedom. With the scales selected, if the curve for the cg response falls below the curve for the hub response, that degree of freedom achieved the 90% isolation criterion; and if the cg curve is above the hub curve, then the 90% isolation criterion was not met.

These plots show that all responses met the 90% isolation criterion with the exception of the cg pitch response to a hub pitch moment at low load levels, although at high load levels the pitch moment also showed well over 90% isolation.

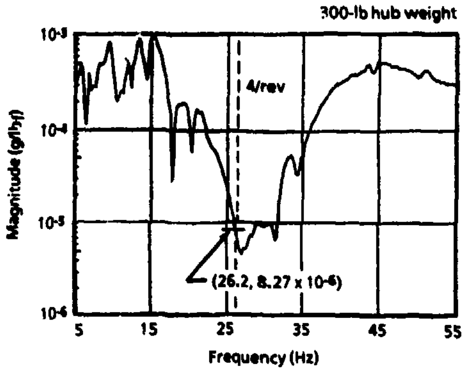

Fig. 6. Pilot seat vertical response to vertical shear force.

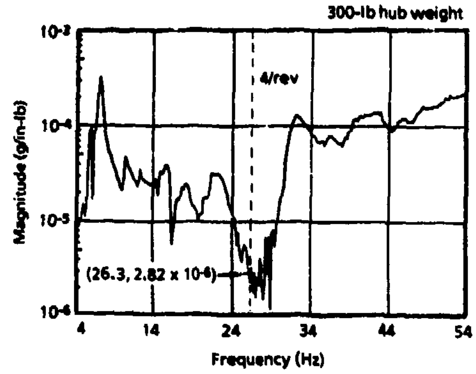

Fig. 7. Pilot seat vertical response to pitching moment.

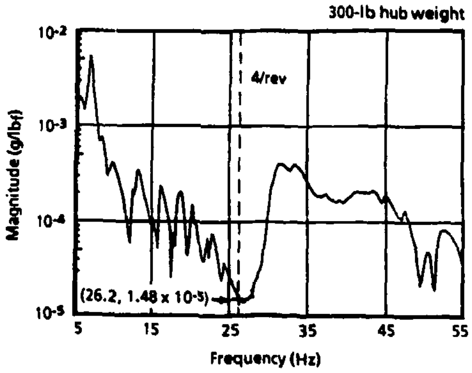

Fig. 8. Pilot seat vertical response to longitudinal shear force.

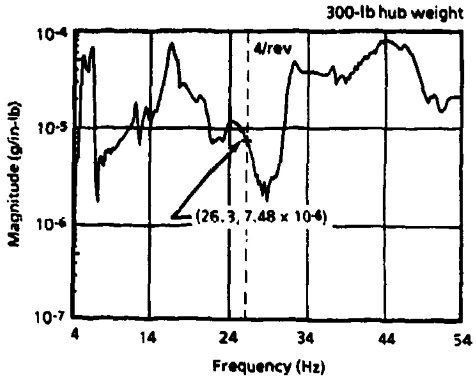

Fig. 9. Pilot seat lateral response to roll moment.

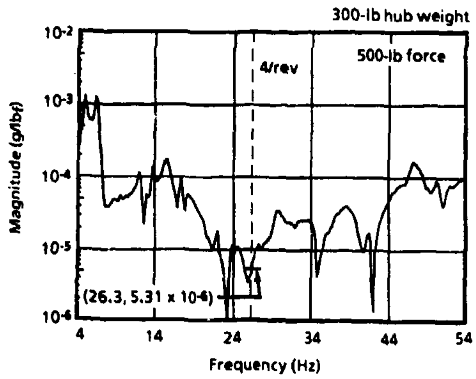

Fig. 10. Pilot seat lateral response to lateral shear force.

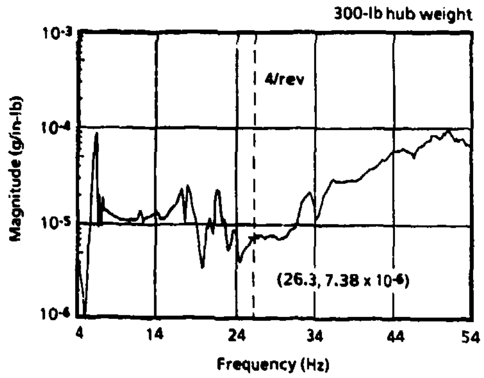

Fig. 11. Pilot seat lateral response to yaw moment.

FLIGHT TEST

Isolation System Performance Flights

Four gross-weight/cg combinations were used during the flight tests:

1. 3500 lb, fuselage station 124
2. 4100 lb, fuselage station 121
3. 3000 lb, fuselage station 127
4. 4100 lb, fuselage station 124

Airspeed sweeps to V_{ne} and various maneuvers were investigated. These maneuvers included right and left turns to 2.5g at 60, 100, and 120 kn; autorotations at 60 and 30 kn; maximum power climbs at 60, 80, and 100 kn; and pushovers and pullups at 60, 80, and 100 kn.

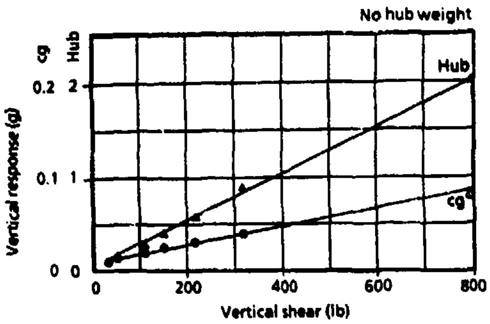

Fig. 12. Fuselage vertical response to vertical hub shear.

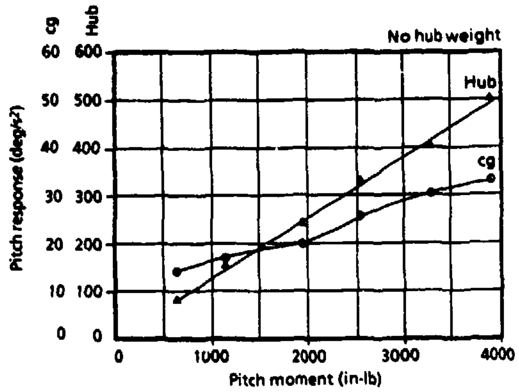

Fig. 13. Fuselage pitch response to pitch hub moment.

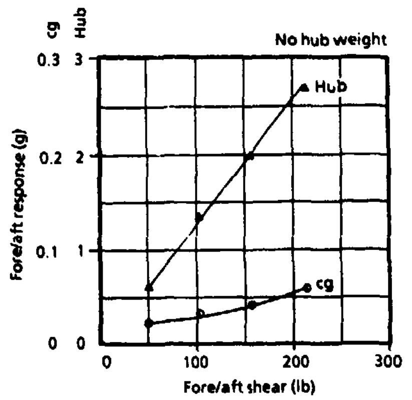

Fig. 14. Fuselage fore/aft response to fore/aft hub shear.

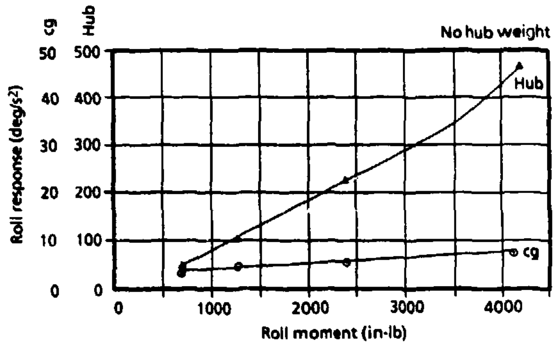

Fig. 15. Fuselage roll response to roll hub moment.

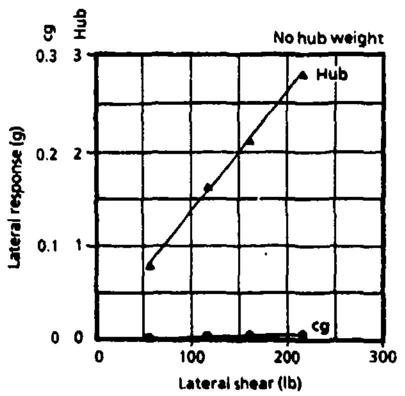

Fig. 16. Fuselage lateral response to lateral hub shear.

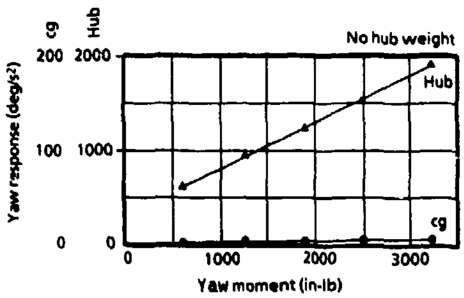

Fig. 17. Fuselage yaw response to yaw hub moment.

Flight Test Results

The results of the flight test are presented in a number of formats. Plots of 4/rev vibration levels at the pilot's seat vs. increasing airspeed in forward flight are presented in Fig. 18. These plots include hover and dive conditions. Plots of 4/rev vibration levels for forward, rearward, right sideward, and left sideward flight up to 30 kn are presented in Fig. 19. Plots of 4/rev vibration levels vs. mean cg g-levels (load factor) for right and left turns, pushups, and pushovers are presented in Fig. 20.

These data show that the pilot's seat 4/rev vibration levels did not exceed 0.075g in any direction during level flight up to 120 kn. This includes all level flight airspeeds in all directions of flight, including the transition region (15 to 25 kn), which traditionally has the highest vibration levels.

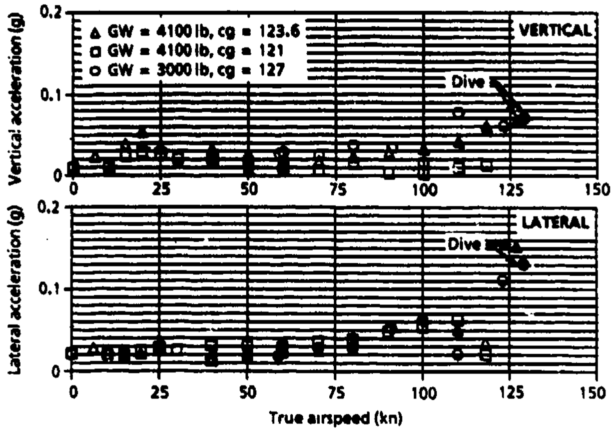

Fig. 18. Pilot seat vertical and lateral acceleration in forward flight.

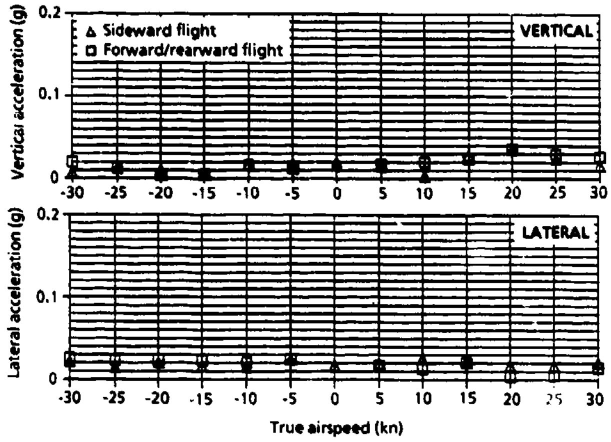

Fig. 19. Pilot seat vertical and lateral acceleration in transition and sideward flight.

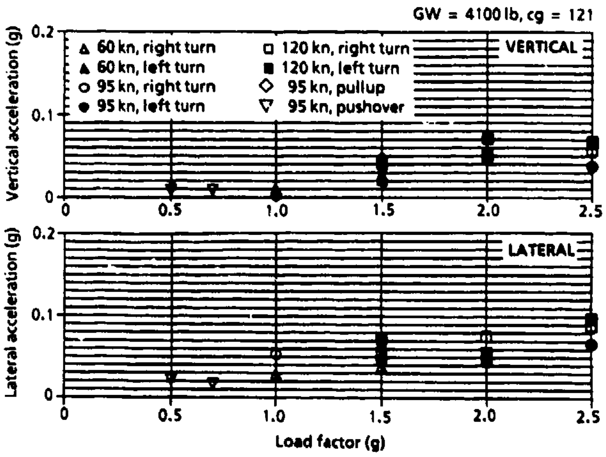

Fig. 20. Pilot seat vertical and lateral acceleration vs load factor.

The plots of 4/rev vibration levels vs. cg mean g-levels for the maneuvers show essentially the same levels at different load factors (mean g) as are achieved at 1.0g, except when the maneuver approaches 2.5g. The TRIS was originally designed for -0.5g to 3.0g. However, during the final tuning phase of the individual LIVE units,[2] the spring rate of the isolators had to be reduced to achieve optimum tuning. This spring rate reduction resulted in two isolators bottoming at approximately 2.5g instead of 3.0g, as initially designed. The two LIVE units that bottomed out at 2.5g were the right forward and left aft units. These two bottomed as a result of the combination of torque and lift, which added as a steady strain in the same direction of these two units. Torque subtracted from lift on the right aft and left forward isolators. The result of this bottoming at 2.5g was an increase in vibration levels at the pilot's seat up to 0.126g. Although these vibration levels were high enough to be perceived by the crew, the levels were still significantly below the levels on the baseline helicopter at the same flight condition. Additionally, this bottoming resulted in no audible sounds to the crew and was only detected by a detailed postflight data analysis.

INDUCED VIBRATION FROM OTHER THAN MAIN ROTOR HUB

Because of the inherent nature of the helicopter, there are other sources of excitation at the main rotor 4/rev frequency than those induced by the main rotor hub loads. These include, first, main rotor downwash onto the cabin roof, elevator, and tail boom, causing vertical vibration; and, second, main rotor downwash onto the elevator endplates, vertical fin, and tail rotor disc, causing lateral vibration. In order to determine the true performance of the pylon TRIS in a flight test program it was necessary to separate the cabin vibrations produced by the main rotor hub loads from those produced by these other sources. This is a difficult task and is beyond the scope of the current program. However, careful study of the shake test and flight test data reveals a reliable measurement of the magnitude and effect of the other sources.

By comparing the level flight airspeed data, it was found that the main rotor hub vibration levels of 1.3g vertically, 1.1g laterally, and 1.0g fore/aft all peak at approximately 20 kn during transitional flight. At this speed, the hub vibration levels are three to four times the levels at the maximum level flight airspeed. Since the transfer functions from the shake test show that the cabin vibration levels are all much less than 10% of the hub g's for all six degrees of freedom, it follows that this relationship would hold true also in a flight test. It can be seen that this is true in transitional flight where hub vibration levels are over 1.0g in all directions, and all the cabin seat accelerometers are under 0.1g, showing better than 90% isolation. At 20 kn, the aircraft is not moving fast enough to cause rotor downwash to impinge on the fin or tail rotor; therefore, no additional cabin vibration levels are produced from airloads on the tail. This is not true at higher airspeeds.

An examination of the V_h and V_{ne} data showed that the hub vibration levels were much lower at high speed than at 20 kn; but this is not true with the crew vibration levels. A review of the vertical and lateral accelerometers on the 90° gearbox revealed the reason: the vibration levels showed a sudden increase that started at about 115 kn and peaked at over 0.5g vertical and almost 1.0g lateral. This sudden increase was caused by the main rotor downwash on the elevator, tail rotor, and fin. The sudden increase in the 90° gearbox lateral vibration caused the pilot's seat lateral acceleration to increase proportionally, and the increase in the 90° gearbox vertical caused the cabin vertical acceleration to increase. Since there are no downwash effects that cause fore/aft vibrations, the cabin cg fore/aft accelerations responded directly and proportionally to the main rotor hub accelerations through the TRIS.

TRIS Performance

It can be seen from the above discussion that the TRIS performance during flight test can only be directly determined in the transition airspeed region where 4/rev excitations from other sources are small. The high-speed cabin vibrations are dominated by excitations from sources other than the pylon TRIS.

In the transition region (shown in the airspeed sweep plots of Fig. 18 and the rearward and sideward flight plots Fig. 19), it can be seen that all the crew accelerometers are below 0.05g for all gross-weight/cg combinations flown, demonstrating that TRIS provided over 95% isolation under these conditions.

BASELINE HELICOPTER COMPARISON

A comparison between the TRIS installed helicopter and the same helicopter with its baseline isolation system was performed. The Model 206LM has been used as a dynamics testbed for many years and has been configured and flown with many isolation systems. The 206LM, however, has never been flown with the transmission rigidly attached to the airframe. For an isolation performance comparison, the Focal Pylon Isolation System was picked.

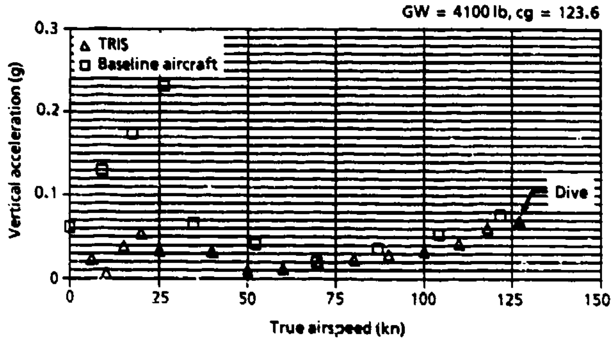

Fig. 21. Pilot seat vertical acceleration with and without TRIS.

The Focal Pylon installation only isolates hub pitch and roll moments, and does not isolate vertical, longitudinal, and lateral hub shears or yaw hub moments. A comparison between the Focal Pylon Isolation System (two degrees of freedom) and the TRIS installation (six degrees of freedom) shows the real potential of TRIS. Fig. 21 shows the comparison of vertical vibrations at the pilot's seat for an airspeed sweep at a gross weight of 4,100 lb and cg of 124 (neutral). These comparisons show a reduction at 20 kn of 75% over the baseline 206LM helicopter vibration levels with the Focal Pylon.

HANDLING QUALITIES

During the initial test flights, the pilot reported the handling characteristics of the TRIS 206LM were significantly improved over any previous 206LM configuration. This improvement was due in part to the standard 206L-1 focused main rotor flight controls installation, which differed from the previous 206LM coupled main rotor control installation. This different control installation resulted in different control inputs to the cyclic and collective controls as they relate to TRIS pylon motion.

Handling qualities evaluations were conducted at both heavy gross weight/forward cg and light gross weight/aft cg. Cyclic and pedal step inputs and pulses were performed during level flight (at 60 kn and 100 kn), descents (60 kn), and climbs (60 kn). In addition, step inputs were conducted in hover. Static lateral directional stability was quantified in level flight (60 kn and 100 kn). Static longitudinal stability was conducted with trim airspeeds of 106 kn in level flight, 60 kn in climb, and finally in autorotation.

Aircraft responses to control step inputs and pulses reflect neutral to slightly positive damping of the longitudinal phugoid at 100 kn and time to double amplitude in excess of 20 seconds at 60 kn at light gross weight/aft cg. Lateral aircraft response to step inputs was generally a slow rolling spiral. Static longitudinal and static lateral directional stick gradients were slightly positive at aft cg. Dihedral was slightly positive at aft cg as well.

The most important improvement was achieved during flight in very gusty air. Because of the control decoupling that was installed into the TRIS pylon isolation system, the rotor responded to gusts with hub moments and shears and deflection of the pylon. Since there was no control coupling to cause the rotor to feather to a new angle (and therefore, some other position in space taking the airframe with it), the dynamics of the pylon/airframe responded like any other transiently excited system and just oscillated about the original position a few cycles until the motion damped out. The results of this effect can be seen in the flight test data plotted in Figs. 22 and 23. For these records, the aircraft was flown in heavy gusts to a stabilized level flight condition, at which point the pilot would tighten the friction on the controls until they would stay fixed with his hands off. The data record was then started and continued until the aircraft finally yawed or rolled off-line excessively, 30 to 40 seconds after the pilot took his hands off. The pilot would then make a control input to level the aircraft. These records were recorded in very gusty air, as seen by the cg load factor trace indicating how much the aircraft was being bounced around. The gusts were strong enough to create pitch, roll, and yaw rates as high as 4 deg/s, but the aircraft stayed within 5° of its initial pitch and roll attitudes even after 40 seconds with no pilot input required. In less gusty air, the pilot has flown for 90 seconds without making control inputs.

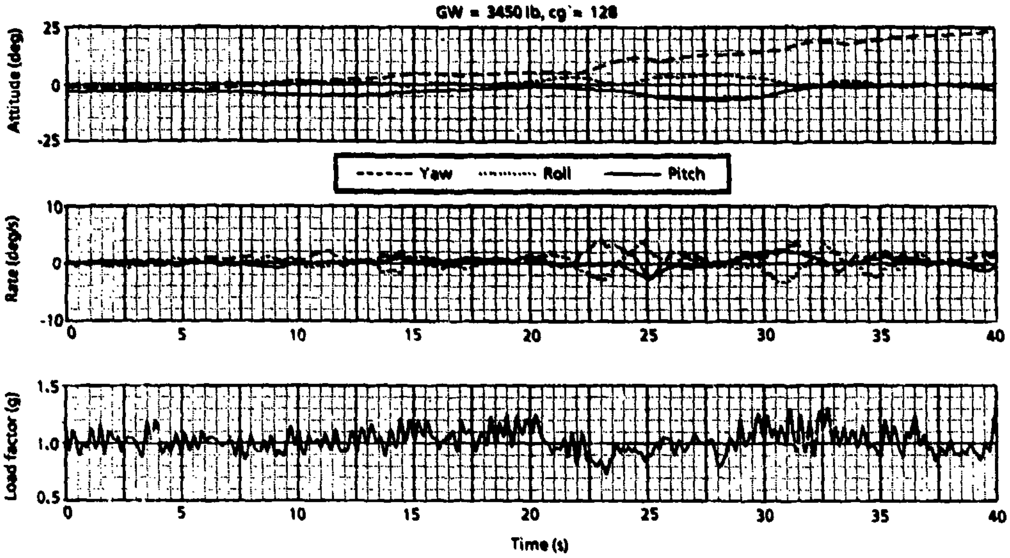

Fig. 22. Damping effect of TRIS during level flight at 80 KIAS through heavy gusts with control friction on and hands off.

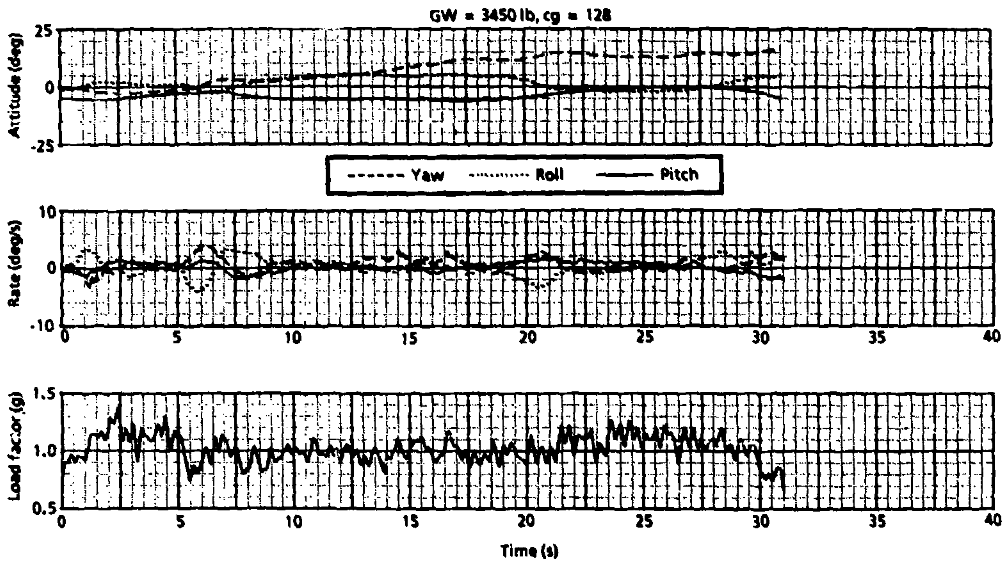

Fig. 23. Damping effect of TRIS during level flight at 100 KIAS through heavy gusts with control friction on and hands off.

CONCLUSIONS

The following conclusions are apparent from the analysis of the ground vibration test and the flight test of the TRIS installation on the Bell 206LM:

1. A six degree-of-freedom pylon isolation system can be made to isolate well over 90% of the main rotor hub loads.

2. The resulting cabin vibration levels (from the remaining 10% of the hub shears and moments that are not isolated) are below perception.

3. There are other n/rev vibration sources that dominate the resulting cabin 4/rev vibration levels and they must be reduced before any additional reductions in cabin vibration levels can be achieved.

4. The highest vibration levels at the crewstations during level flight to 120 kn measured less than 0.1g and were imperceptible to the crew. The justification for a military criterion less than this level must be questioned.

5. The TRIS installation had a weight penalty of 69.57 lb on the 206LM.[2] This is less than 1.75% of the maximum gross weight of 4,100 lb. This installation was designed to be adjustable and therefore somewhat heavier than a production system need be. By manufacturing the LIVE units without adjustability and out of lightweight material (stainless steel was used for this proof-of-concept test,, less than 1% weight penalty is easily achievable.

6. The objective of this flight test program (all six degrees of freedom isolated over 90% in flight) has been met with the TRIS installation. However, the desired goal of less than 0.05g throughout the level flight envelope was not met because of other airframe excitations that dominate the remaining vibrations at high speed.

7. A comparison of the TRIS vibration levels to the Focal Pylon vibration levels do not show the maximum potential of a TRIS. If the comparisons were made to a rigidly mounted transmission, an even greater reduction from baseline would be observed.

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MINIMISATION OF HELICOPTER VIBRATION THROUGH ACTIVE CONTROL OF STRUCTURAL RESPONSE

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SUMMARY

Vibration still remains as one of the major problem areas for rotary winged flight. In order to control helicopter vibration, many absorption and isolation schemes have been applied with limited success. Advances in both computational techniques and actuator technologies have created a firm base for the development and application of active vibration control strategies to the helicopter.

This paper details one such technique and its current demonstration installation on a Westland 30 helicopter. Termed Active Control of Structural Response (ACSR), the technique employs high-frequency force-actuation within the helicopter's structure. These forces are superposed onto the dominant vibratory forces in an active manner, such that the fuselage vibratory response is minimised. The basic control philosophy for ACSR is described and the current experimental demonstration status is detailed.

1. INTRODUCTION

The control of vibration has been and remains, a problem for all rotary winged vehicles. Considerable efforts have been expended over many years in attempts to reduce vibration to acceptable levels. On the helicopter there are many sources of vibration, but the most important component is generated by the main rotor and occurs at a frequency (bR) equal to the product of the number of blades (b) and the rotor speed (R). This blade passing frequency vibration is an inherent consequence of driving a rotor edgewise through the air, and can never be completely eliminated, although the magnitude of the rotor excitation can be controlled by careful rotor system design. The response of the airframe is also sensitive to the dynamic characteristics of the fuselage, and again careful design can minimise the response. As understanding of the nature of the problem has increased, and the ability to predict the dynamic response of both rotor and airframe has improved, it has become possible to design a helicopter for low vibration, or at the very least to avoid those problems which have led to very high vibration in the past. The trend for increased cruise speed, and mission endurance has, however, aggravated the problem, since the magnitude of the rotor vibratory loads increases with speed, and the effect of vibration on human fatigue is proportional to exposure time.

Over the years a number of passive methods for tackling the problem have been developed, indeed helicopter dynamicists are renowned for their ingenuity. Since vibration originates as airloads acting on the rotor blades then it is not suprising that many features of the rotor are important; these include the number of blades, type of hub, frequency placement with respect to rotor speed of the blade modes and coupling between flapping and torsion in the modes. However, the rotor design process is necessarily a compromise between a number of conflicting requirements, and some residual forcing will always be present.

Having achieved the best possible rotor behaviour it is necessary next to consider the response of the airframe. This can be controlled by designing the airframe to be non-resonant; isolating sensitive areas of the airframe from the source of excitation, and by fitting vibration absorbers. Non-resonant airframe design is now standard practice throughout the industry. However, studies have shown that the rotor loads are often enough to give unacceptable vibration even if the fuselage behaved as a rigid body. Thus it is not sufficient for the fuselage to be non-resonant, it must be anti-resonant and this is impossible to achieve throughout the airframe.

Vibration absorbers have proved very successful in the past, but always have the disadvantage of involving some weight penalty, and in the case of rotor head mounted absorbers the weight penalty can be considerable, say 1% of the gross aircraft weight. Rotor head mounted absorbers also increase the aircraft drag. Nevertheless this is a preferred mounting position as it is close to the source of vibration and consequently, this form of vibration control is often adopted.

Fuselage isolation systems, based upon either simple soft springs or more complex anti-resonant principles (eg. DAVI's) have been tried on a number of aircraft, with mixed success. Such systems are usually mounted at the fuselage to main gearbox interface. One considerable problem is that until now such systems have been passive, and therefore produce loads which are related only to the displacements at the attachment points. This means that any non-isolated load paths connecting the rotor to the fuselage can seriously degrade the systems performance.

Other problems with anti-resonant systems are the effect of variations in rotor speed on attenuation, and the maintenance requirements to keep the system tuned. Simple soft-mounting systems do not suffer from this problem, but on the other hand it is difficult, if not impossible, to obtain a sufficient degree of attenuation without using unacceptably soft springs.

With the advent of increased computing power the use of active techniques to control vibration has become a real possibility. One favoured and much studied method is to oscillate the blades in pitch at harmonics greater than the first (known as Higher Harmonic Control or HHC) in an attempt to reduce the loads transmitted to the airframe. Trials by a number of companies have shown very promising results, at least for flight well within the rotor performance envelope. Furthermore, studies at Westland have indicated reductions of typically 80 - 90% in airframe vibration. The need for an active (self-adaptive) control system arises from the complex nature of the rotor loads and the fuselage dynamics. Non-adaptive systems are unlikely to be successful.

A number of potential difficulties exist with the implementation of HHC. These include a possible degradation of rotor performance, due to conflict between retreating blade stall at high forward speeds and the requirements of vibration reduction. Care must also be taken with the airworthiness implications of modifying the primary flight control circuit, which may require stiffening to improve the blade response to higher harmonic actuator inputs.

In this paper a new approach to helicopter vibration control is considered. We believe it combines the best aspects of soft or anti-resonant isolation systems with modern active control technology, and avoids the potential rotor performance and airworthiness problems associated with HHC. The new approach is the Active Control of Structural Response (ACSR). The theoretical basis of the technique is described in the next section. The principle consists of connecting a number of actuators between convenient points on the airframe and applying forces to the structure through these actuators. The magnitude and phase of the loads generated by the actuators are chosen to minimise vibration at a number of locations in the fuselage, with the system being controlled by an active control algorithm.

The basis of the technique is given in the following sections, followed by a study of the potential benefits and application of ACSR as applied to the Westland 30/Series 100. Finally, the advantages and disadvantages of ACSR are compared with those for HHC.

2. ACTIVE CONTROL OF STRUCTURAL RESPONSE - BASIC THEORY

In this section the basic theory behind the active control of structural response is developed. Consider any linear dynamical system, and let the normal modes of the system be ϕ_n, with corresponding natural frequency w_n, modal mass m_n and ratio of critical damping z_n. The suffix n runs over the range 1 to N, where N is the number of normal modes.

Suppose the system is acted upon by a set of external oscillatory loads, all at a single frequency W. In the helicopter case these loads are the rotor generated vibratory hub forces. If $\underline{F}_R$ is defined as the vector of external loads applied at point R, then in Euler form the sinusoidal forcing is defined as:

$$\underline{F} = \underline{F}_R\, e^{iWt} \tag{1}$$

where the individual elements of $\underline{F}_R$ are complex numbers and i is defined as the square root of -1. Then in the usual way, the response of the structure to the rotor forces alone in physical co-ordinates is given by:

$$\underline{Y} = \sum_{n=1}^{N} \frac{\underline{\phi}_n \left(\underline{\phi}_n^T (R)\, \underline{F}_R \right)}{m_n \left(w_n^2 - W^2 + 2\, i\, z_n w_n W \right)} = \underline{B} \quad \text{say} \tag{2}$$

where $\underline{Y}$ is the vector of displacements at each point in the system, $\underline{B}$ denotes the background vibration vector at each point in the system and T denotes the vector transpose.

Suppose now that a force U is applied to the structure at point P and an equal and opposite force is applied at point Q, where:

$$U = X_1\, e^{iWt} \tag{3}$$

Then the vibratory response of the system, Y, to this force is given by:

$$\underline{Y} = \sum_{n=1}^{N} \frac{\underline{\phi}_n \left(\phi_n (P) - \phi_n (Q) \right) X_1}{m_n \left(w_n^2 - W^2 + 2\, i\, z_n w_n W \right)} = \underline{T}_1\, X_1 \tag{4}$$

where T_1 is the respon... of the system to a unit force X_1 applied between points P and Q and $\emptyset$ (P) etc denotes the value of $\emptyset$ at point P.

Combining the response to the external force with that generated by force X_1, equations (2) and (3), the response becomes

$$\underline{Y} = \underline{T}_1 X_1 + \underline{B} \tag{5}$$

and by choosing

$$X_1 = - B(S)/T_1(S) \tag{6}$$

the response $\underline{Y}$ at any chosen point in the structure S can be made zero, provided $T_1(S)$ is non-zero. Thus, with a single control force the response at a single point in the structure may be reduced to zero.

Generalising to K force actuation points. the total response of the structure becomes:

$$\underline{Y} = \sum_{j=1}^{K} \underline{T}_j X_j + \underline{B} = [T]\,\underline{X} + \underline{B} \tag{7}$$

where [T] is a matrix with columns T_j, and $\underline{X}$ is a vector of the control forces X_j. Equation (7) is the basis of ACSR. In general with K control forces, the responses at K separate locations in the structure may be reduced to zero, provided the sub-matrix of [T] relating to the K response points to the actuator loads is non-singular.

Blade passing frequency vibration in the helicopter is typically in the range of 10 to 30 Hz, and the fuselage response will be dominated by the modes in the range from zero to twice blade passage frequency. Looking again then at equation (4) it is clear that if

$$\emptyset_j(P) = \emptyset_j(Q)$$

in all the modes in the frequency range of interest the response at S due to X_1 will be zero. Thus to be successful it is important that there is some relative motion between the points P and Q in the modes to be controlled. Furthermore, the larger this relative motion the lower the required force needs to be. The Westland 30 series 100 aircraft is ideal in this respect, since it already incorporates a soft transmission mounting system across which the actuators can be attached.

In order to implement the technique a control strategy is required which schedules the actuator magnitudes and phases such as to minimise fuselage vibration. A discussion of the control options and the selected scheme for flight demonstration is given in the following section.

3. CONTROL STRATEGIES

Broadly, two categories of control algorithm are applicable to the implementation of ACSR, which may be classified as either frequency or time domain in nature.

The time domain algorithms are based on the continuous feedback of vibration through a set of pre-determined control gains. There are a number of potential difficulties with the implementation of these types of strategy. Primarily, the selection of suitable gains for vibration minimisation depends on accurate knowledge of the helicopter's dynamics. In practice, the dynamics are ill-defined and will change with aircraft all-up-weight. A number of design techniques have been developed for generating control laws for multi-dimensional flexible structures. The most promising of these techniques is Independent Modal Space Control(IMSC). This technique hinges on the ability to control a number of vibratory modes through the feedback of modal information. In general, the application of this control law requires one actuator per controlled mode. However, a sub-optimal version of IMSC has been developed (reference 1), whereby the restriction of one actuator per mode is relaxed. Given that the number of actuators is significantly less than the number of modelled modes, then it is necessary to use a sub-optimal formulation to determine the actuator commands which yield the required generalised forcings in each mode. Control over the way in which vibration is minimised is achieved through the relative weighting of the dominant modes. The feedback of modal information does, however, dictate the need for 'observers', whose function is to estimate modal information based on the measured vibration. Essentially, the derivation of control gains is based on the minimisation of a quadratic performance index. The object of this index is to directly minimise the structure's modal response without excessive control effort. Minimisation of this index gives a feedback formula for each modelled structural mode in terms of the generalised forcing required for that mode. Finally, it is necessary to determine the actuator commands which realise these generalised forces. This approach attempts to increase the structure's natural damping and also allows the modal frequencies to be moved. Consequently, at its best this approach will make the fuselage non-resonant rather than anti-resonant. The major advantage of this technique is that it allows for the minimisation of vibration over a wide frequency range.

Frequency domain control algorithms are based on the minimisation of vibration at the dominant blade passing frequency alone. The assumption is made of a quasi-static linear relationship between the measured vibrations resulting from a given set of force inputs, at blade passing frequency, of the form given in equation (7). To maintain this steady linear relationship, it is necessary to use high resolution Discrete Fourier Transforms (DFT's) over an appropriate time interval to extract the fuselage forced response from the decaying transient vibration. The minimisation of vibration is achieved through the optimisation of a performance function J, defined as:

$$J = \underline{Y}^T [C]\,\underline{Y} + \underline{X}^T [D]\,\underline{X} \tag{8}$$

where [C] and [D] are diagonal matrices of weighting factors. The weighting matrix [C] allows for the possibility of certain fuselage locations being more important with respect to vibration than others. The term $\underline{X}^T [D] \underline{X}$ is included to allow limiting of the actuator commands within their practical constraints. Minimisation of this performance function yields the following optimal control formulation:

$$\underline{X} = - \left([T]^T [C] [T] + [D] \right)^{-1} [T]^T [C] \underline{Z} \qquad (9)$$

This approach is identical to that proposed for Higher Harmonic Control (HHC) as detailed in Reference 2. For HHC, the coefficients of the assumed linear relationship are known to vary with flight condition and hence, statistical estimators are used to track them during flight. However, unlike HHC the transfer relationship for ACSR can be assumed to be constant for given fuselage dynamics. This allows the use of a number of options which depend on either the estimation of all the background parameters (of the assumed linear relationship) or only a limited number of parameters.

In order to proceed with the design of a vibration controller, it was necessary to make an early decision on the type of control strategy for the demonstration of this technique. Through extensive simulation work it was concluded that the time domain options were difficult to realise and did not prove robust to changes in aircraft dynamics. Furthermore, the degree of vibration minimisation was not as good as for the frequency domain algorithms. Also, the frequency domain methods proved both to be robust and simple to realise, although they are significantly more complex to implement in controller form.

Early work at Westland (funded by the RAE) on the development of a multivariable controller for HHC had developed the controller architecture to a reasonable degree of sophistication. Hence, it was decided to ruggedise this hardware and develop the frequency domain algorithms for flight demonstration on the Westland 30.

The general arrangement for the frequency domain control strategies is shown in Figure 1. This indicates that the primary controller functions are signal processing, parameter estimation and optimal control.

In order to demonstrate ACSR a number of controller options have been implemented in microprocessor form. These options are summarised below in Table I.

OPTIMAL CONTROLLER TYPE		ESTIMATOR TYPE
MODEL TYPE	STRATEGY	
Global	Deterministic	Kalman Filter Recursive Least Squares
	Stochastic	Kalman Filter Recursive Least Squares
Local - Linear	Deterministic	Kalman Filter Recursive Least Squares
	Stochastic	Kalman Filter Recursive Least Squares
	Reduced	None

TABLE I Control Algorithm Options

These options arise from four basic considerations.

(a) The background linear vibration model can be assumed to be either global or incremental in nature. This gives rise to the Global and Local-linear representations.

(b) The optimal controller, which provides the minimising actuator commands can be either deterministic or stochastic. The deterministic controller assumes perfect knowledge of the background parameters. The stochastic controller considers the fact that the parameters may be estimated. This has the effect of making the controller 'cautious'.

(c) A further consideration arises from the assumptions about the background parameters. The full (HHC type) controller assumes that these parameters vary with flight condition and hence, estimators are used to track these changes. However, given that for a particular set of helicopter dynamics the transfer matrix [T] remains constant, then a reduced statistical estimator can be used.

(d) Given that a full estimation strategy is adopted, then two basic forms of estimator are applicable, the Kalman filter or the Recursive Least Squares estimator (with variable forgetting factor).

One of the main problems with the use of the frequency domain technique relates to the assumption of a static linear background model. This assumption implies that it is necessary to extract the steady forced response from the transient response. In practice, information from this transient period is used by the signal processor to help identify the static forced vibration response. The timing diagram for the controller's operation (Figure 2) illustrates the control sequence. The fuselage transient response is of the order of 2% critically damped and this level of damping will give transient decay times in excess of 3 seconds. However, the use of DFT's requires a signal processing time of approximately three rotor revolutions to identify the blade passing frequency (22 hz for the Westland 30) forced vibration response from the surrounding transients (which occur at the structure's natural frequencies). In this instance, the frequency resolution of the filter is 22 Hz +/- 1.5 Hz.

The implementation of the Frequency Domain algorithms on an adaptive controller has shown update rates to be of the order of one second (this comprises 600 ms for signal processing and 400 ms for algorithm calculations).

The algorithm performance is significantly affected by controller update rates. Given steady aircraft flight conditions then the typical algorithm performance is characterised by a 4 - 6 second period during which the transient vibrations decay, (Figure 3). This transient period is often accompanied by rapid fluctuations in control action while the parameter estimator adjusts to more optimal conditions. The steady state performance can be improved by the effective limiting of the rate of change of the actuator inputs through using the stochastic or 'cautious' controllers which will give a smooth transient period (Figure 4). A reduction in the transient period can be achieved by using correct initial estimates for the dynamics.

The performance of the various controller configurations using the one second update has been evaluated for a variety of manoeuvre conditions. It is apparent that using this update rate, the vibration reduction capability during fast manoeuvres is limited. Figure 5 shows the effect of a fast ramp velocity manoeuvre (119-133 knots in 2 seconds). The results indicate that the filter detects the manoeuvre and attempts to retrack the parameter estimates. This period of change is characterised by rapid fluctuations in the Kalman gain, which represents the amount by which the estimates are updated. The problems with tracking manoeuvres are directly attributable to the slow update time.

During the above ramp there are only two filter update cycles. This does not allow the filter to adjust to the change. Consequently, there is a further transient period after the ramp where the vibrations settle. The performance of the controller during slower manoeuvres is significantly improved. Generally, the filter requires at least 5 or more cycles to adjust to the change. For example, the filter is able to adapt and continue to minimise vibration for manoeuvres of the order of 2 knots/second.

4. ACSR ON THE WESTLAND 30

ACSR is particularly suitable to the Westland 30, whose raft construction (Figure 6) allows force actuators to be incorporated into the gearbox/fuselage interface at the four elastomeric mount locations. The technique in this application is often termed Active Gearbox Interface Control (AGIC). Westland are currently engaged on a programme of work leading to a proof-of-concept demonstration on a Westland 30 through flight testing in late 1986. This initial phase of research and flight demonstration is sponsored by the Royal Aircraft Establishment (RAE) and the Ministry of Defence/Directorate of Future Systems (MoD/DFS).

The basic control system comprises ten airframe vibration sensors (see section 4.5), an active controller and four force-producing electro-hydraulic actuators. In addition to the above the full experimental installation includes a flight engineer's station, pilot's panel and data recording facilities. A schematic of the proposed system is given in Figure 7, whose major system elements are described as follows:

4.1 Adaptive Control Unit

This is a multi-processor based adaptive controller whose primary function is to schedule the magnitude and phases of the four actuator force inputs such that airframe vibration is minimised. In addition to the vibration control algorithm, the following functions are implemented within the controller:

(a) Signal Processing - decomposes measured vibration into 4R vibration components using Discrete Fourier Transform methods.

(b) Actuator Servo-Loop Control - this provides the servo-actuator drive signal and incorporates two actuator servo-loop control laws. The primary loop is based on force feedback and provides force-following, ensuring that the actuator sees no mean load. The secondary control law arises due to the possible coupling of the actuators across the raft, and incorporates differential velocity and acceleration feedback from across the raft fuselage interface.

(c) Executive Processor - controls the interface to the flight engineer's station and provides information for a real-time graphics display.

(d) Data Acquisition System Interface - loads required instrumentation data into buffers, which are accessed by a Modular Data Acquisition System (MODAS).

4.2 Active Elastomers

A force producing actuator is incorporated within each of the four modified elastomeric mounts (Figure 8). The function of the actuators is to inject forces into the structure at blade passing frequency, but without any resistance to the low-frequency raft motions.

4.3 Engineer's Station

This consists of a ruggedised computer monitor and keyboard. The monitor will provide a real-time graphics facility, displaying vibration, force and algorithm information. Provision will be made for all necessary tests to be carried out by the flight engineer.

4.4 Pilot's Panel

Primarily the pilot's panel gives the pilot authority over the control system. Warning indicators for the hydraulic supply system and an actuator by-pass indicator are included. The pilot is able to switch the hydraulic isolation solenoid and provide the engineer's station with the authority to take the actuators out of by-pass. Also, the pilot can disable the control system at any time, thus returning the helicopter to its original state.

4.5 Vibration Sensors

Twenty-four vibration locations throughout the fuselage will be linked to the controller. These twenty-four locations will be divided into two sets of twelve, from which the flight engineer can select five from each twelve. This allows some flexibility of the sensors used by the vibration controller and will allow for system optimisation.

4.6 Data Acquisition System

Use will be made of MODAS for flight data recording. Both sensor information and digital algorithm parameters are to be recorded for later analysis.

5. THE BENEFITS OF ACSR FOR THE WESTLAND 30 SERIES 100

As described previously the Westland 30 is fitted with four elastomeric mounts, which are relatively soft in the vertical and fore-aft directions, and stiff in the lateral direction. The aircraft is also fitted with a rotor mounted mass-spring absorber, and the combination of soft-mounted raft and absorber produces an aircraft with levels of vibration typical of the modern helicopter. Without the absorber the levels of vibration, especially at the front of the aircraft, are high (see Reference 3).

This section discusses predicted vibration performance of ACSR on the Westland 30, assuming perfect knowledge of the aircraft dynamics. The control system is based on that previously described, using twelve accelerometers attached to the floor of the cabin; ten measuring vertical vibration and one each for cockpit lateral and longitudinal vibration. The consequences of using a much reduced set of accelerometers is also discussed.

The mathematical model used for this study is a NASTRAN finite element model which has been validated against the aircraft. The rotor head absorber is not included in these simulations of the ACSR system. In Figure 9 the predicted vibratory response at blade passing frequency at six of the twelve monitoring positions (with and without the ACSR system working) are compared with flight measurements on the aircraft. The flight measurements were obtained with the absorber fitted to the aircraft; flight measurements without the absorber agree closely with the predictions. It is clear form Figure 9 that not only does ACSR produce a very substantial reduction in vibration, but also it works much better than the dynamic absorber. The results for the other six locations are very similar to those shown.

It might be thought that with the actuators positioned across the interface between the gearbox and fuselage that the gearbox vibration would be increased by ACSR, but this is not so. The vibration at all positions on the main gearbox and raft is also reduced, by a similar magnitude to that predicted for the cabin, even though these points are not included in the cost function of airframe vibration. The reason for this is that as configured on the Westland 30, ACSR is not an isolation system, but a modal cancellation system, and the effect has been to greatly reduce the response of the complete aircraft without ACSR. Indeed, it is the self-same modes which the rotor head absorber attempts, quite successfully, to control, but clearly ACSR is superior to the absorber.

If only four sensor locations are used to control the active system then with four actuators the vibration at those four points can be reduced to zero. However, the levels of the other, un-monitored, positions is not reduced as much. This result is summarised in Figure 10 where the average vibration at the twelve monitor postions without ACSR is compared with the levels predicted for twelve and four position control. The slight degradation in overall performance with only four monitor points can be seen, but the change is slight. For the flight trials of ACSR only ten locations are being controlled, since other studies have shown this to be adequate. Overall, ACSR is predicted to reduce vibration by about 90%.

The forces required to achieve the above reduction in vibration are shown in Figure 11 for both twelve and four point monitoring. The significantly lower force levels required with the higher number of monitoring positions is useful with regard to sizing the actuators and hydraulic system.

6. COMPARISON OF ACSR AND HHC

It is instructive to compare the active control of structural response with higher harmonic blade pitch control, and in this respect a number of issues are important. These include the degree of vibration reduction, effect on aircraft performance, airworthiness issues, installation difficulties and general applicability.

6.1 Vibration Reduction

The performance of HHC depends to some extent upon whether the actuators are mounted in the fixed frame (in series with or part of the flight control system, below the swashplate) or in the rotating frame (as part of the rotor or hub). For a four bladed semi-rigid rotor (similar to that on the Westland 30

series 100) studies carried out at Westlands have shown reductions of vibration of 90% with fixed frame actuation, and 95% with rotating frame actuation. The advantage of rotating frame actuation on a four bladed rotor is that it produces 2R blade pitch motion.

Since the predicted vibration reduction of ACSR is also 90% it would appear that HHC is only slightly better than ACSR, and then only if rotating frame actuation is adopted. The predictions of vibration reduction for HHC do not include any aerodynamic constraints on rotor performance.

6.2 Rotor Performance

The study of HHC has shown that its use results in some increase in rotor bending moments and a reduction in blade stall margins. Indeed, at Vne the stall margin is zero, consequently any HHC inputs would penetrate stall, and significant overblading would be required to restore the stall margins necessary to optimise HHC. The HHC inputs also increase the oscillatory pitch link loads.

ACSR will have no effect on rotor stresses, stall margins or control loads.

6.3 Airworthiness

By its very nature fixed frame HHC will have some impact on the primary helicopter flight control system. Any failures of the HHC system must still allow the safe operation of the helicopter. Individual blade actuation, by tip tabs for example, would overcome some of the control system airworthienss problems, but introduce others (for example tab flutter).

Because the ACSR system consists of hydraulic jacks in parallel with the existing elastomers, minimal airworthiness issues are involved.

6.4 Installation Difficulties

One of the possible problems with fixed frame HHC actuation is the lost motion between the jacks and the blade torsional displacement. To avoid this problem considerable redesign of the controls might be required. No such problems should be encountered with ACSR.

6.5 General Applicability

Clearly, HHC should be applicable to any helicopter; all that is required is space to fit the actuators. ACSR is also generally applicable, since all it requires is points on the aircraft which have relative motion at blade passing frequency. However, finding the ideal locations will not necessarily be easy, unless the airframe already incorporates a significant area of flexibility, as is the case on the Westland 30.

A further advantage of HHC is that it may be incorporated as part of a general ACT system, and therefore the weight penalty could be less than with ACSR.

Provided the predictions for ACSR are confirmed by the flight trials, which should take place shortly, then we believe that overall ASCR has more advantages than HHC.

7. CONCLUSIONS

The technique of Active Control of Structural Response has been shown through extnesive research studies to have significant potential for helicopter vibration reduction. Predictions of fuselage vibratory response are far superior to passive systems and compare favourably to other active techniques, such as Higher Harmonic Control. Furthermore ACSR is both generally applicable to any structure and has minimal associated airworthiness issues.

Control algorithms for the implementation of ACSR have been fully developed and tested on a flightworthy adaptive controller.

A demonstrator programme is now underway based on a Westland 30 series 100 helicopter. The programme of work is mature and it is expected to culminate in a flight evaluation later this year. A two phase test programme is to be undertaken, consisting of shake tests to evaluate system operation and optimise algorithm performance, followed by a limited flight test programme.

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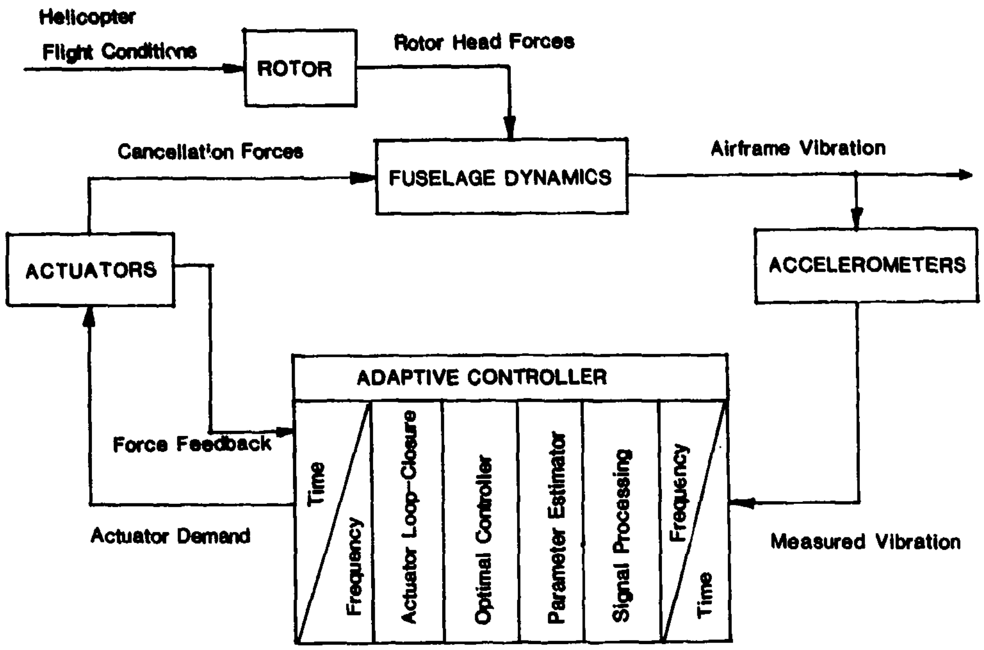


FIGURE 1. Active Control of Structural Response Schematic

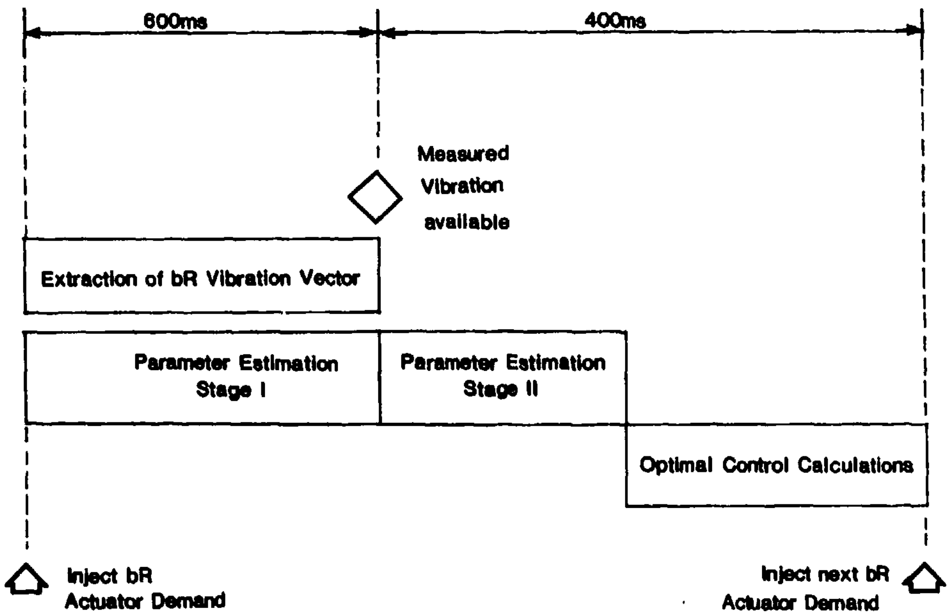
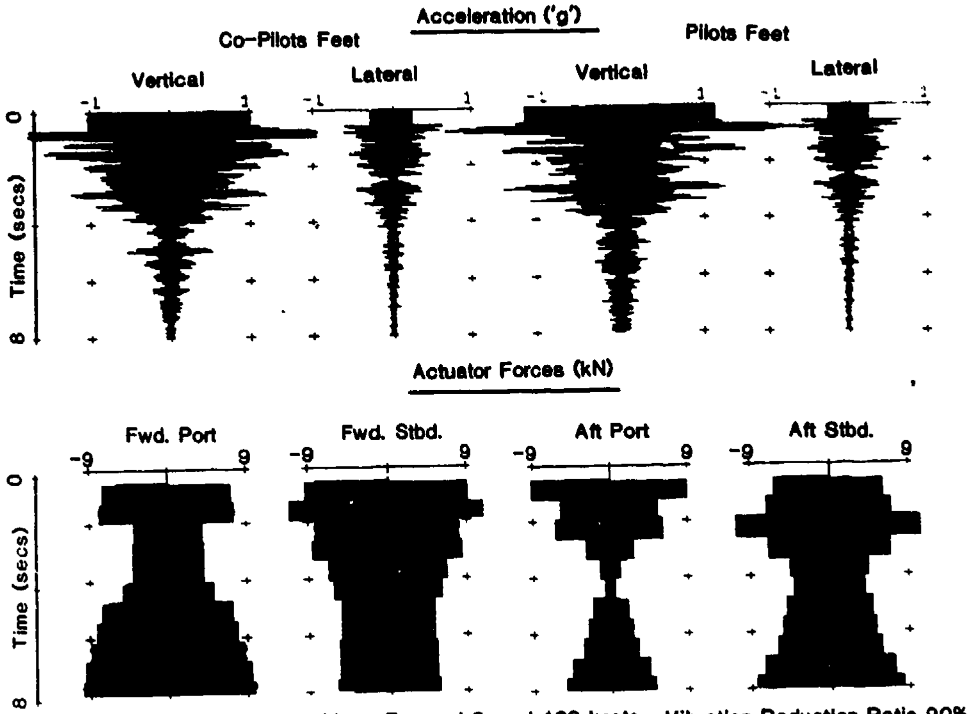
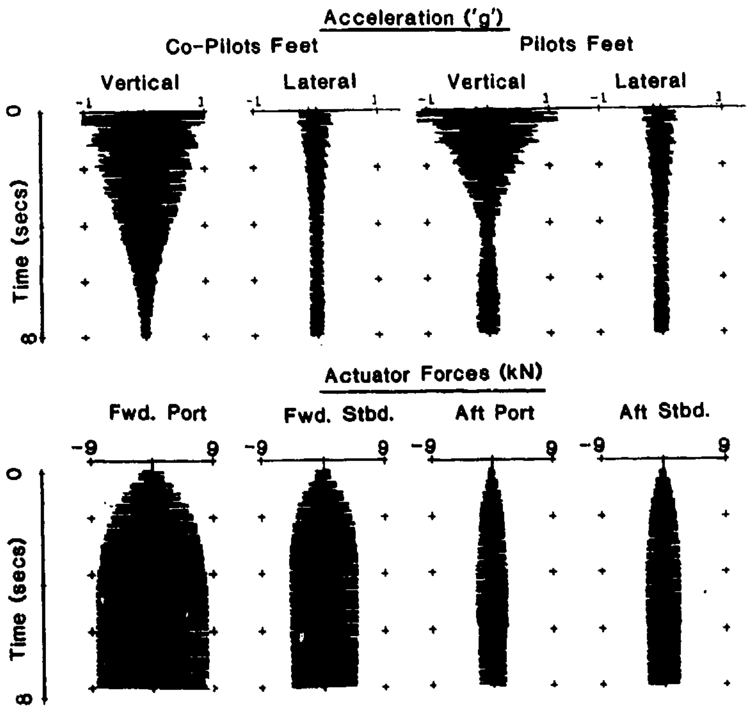


FIGURE 2. ACSR Vibration Control Sequence



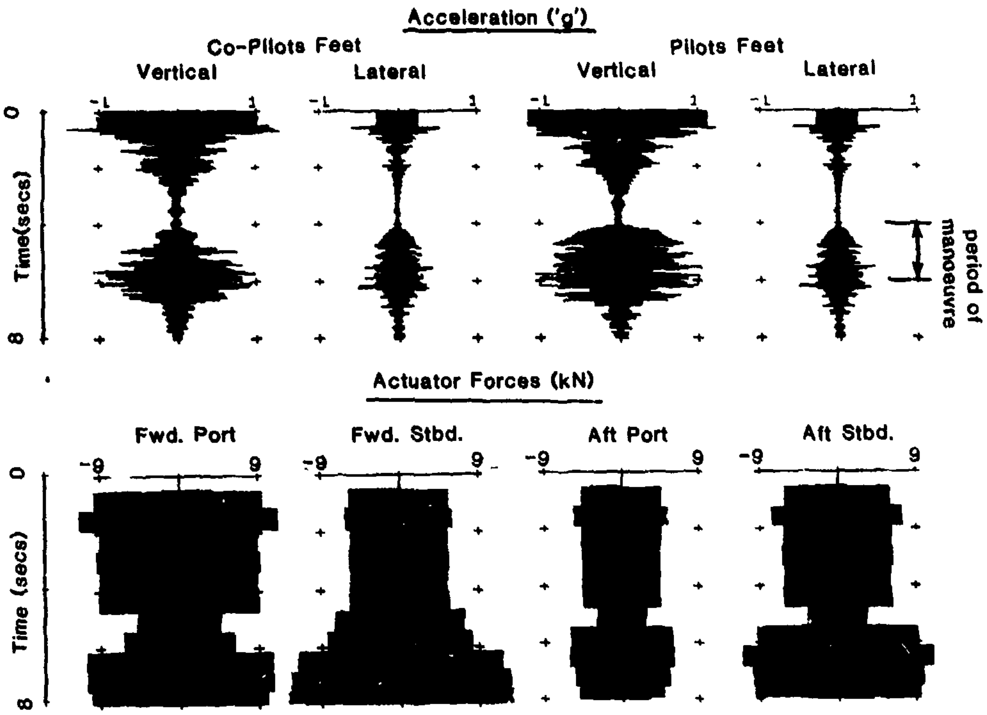
Global Deterministic Algorithm, Forward Speed 120 knots, Vibration Reduction Ratio 90%

FIGURE 3. Vibration Control Algorithm Performance - Steady Flight Conditions



Local-Linear Stochastic Algorithm, Forward Speed 120 knots

FIGURE 4. Vibration Control Algorithm Performance - Steady Flight Conditions



Global Deterministic Algorithm, Manoeuvre Condition 120 -'130 kts

FIGURE 5. Vibration Control Algorithm Performance - Manoeuvre Response

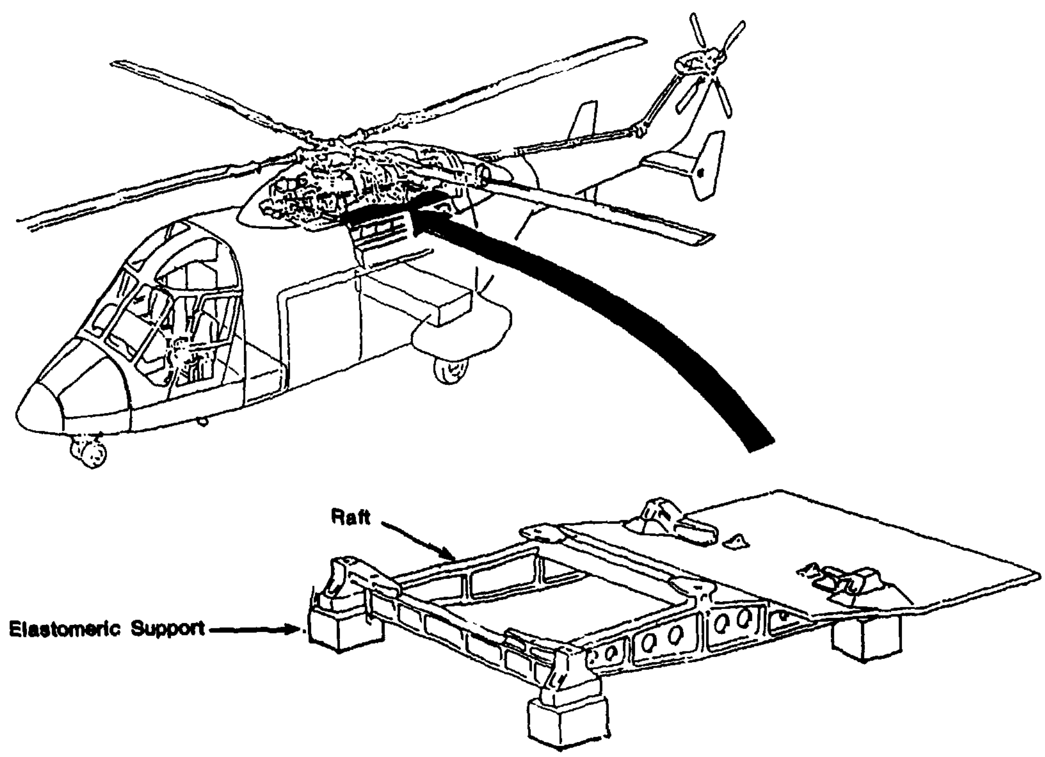


FIGURE 6. Westland 30 Series 100 Raft Construction and Location

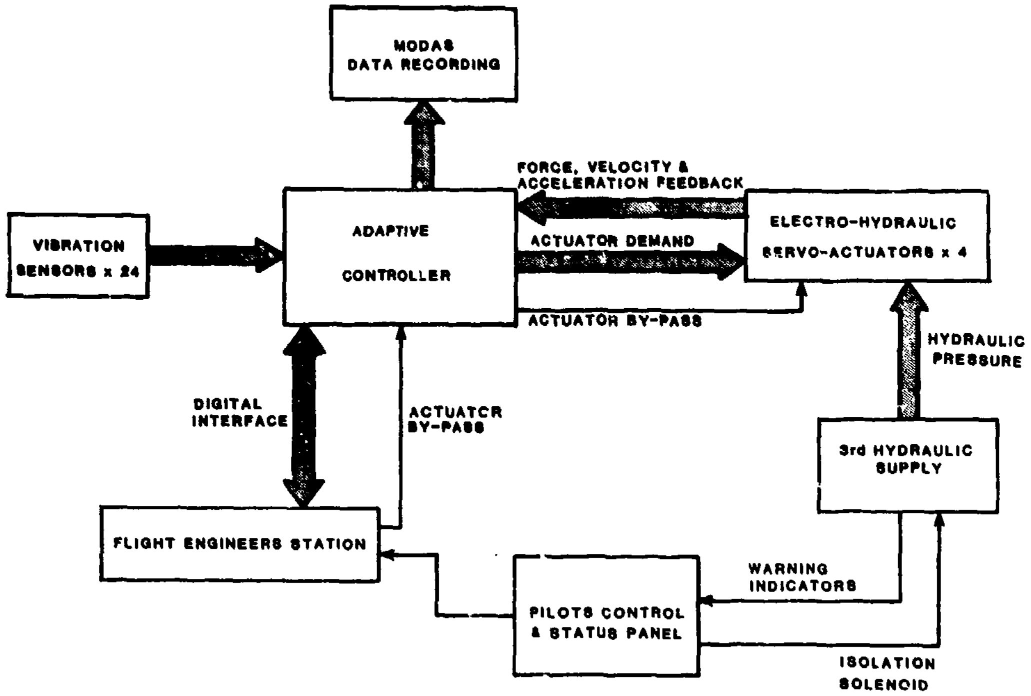


FIGURE 7. Active Control System Installation

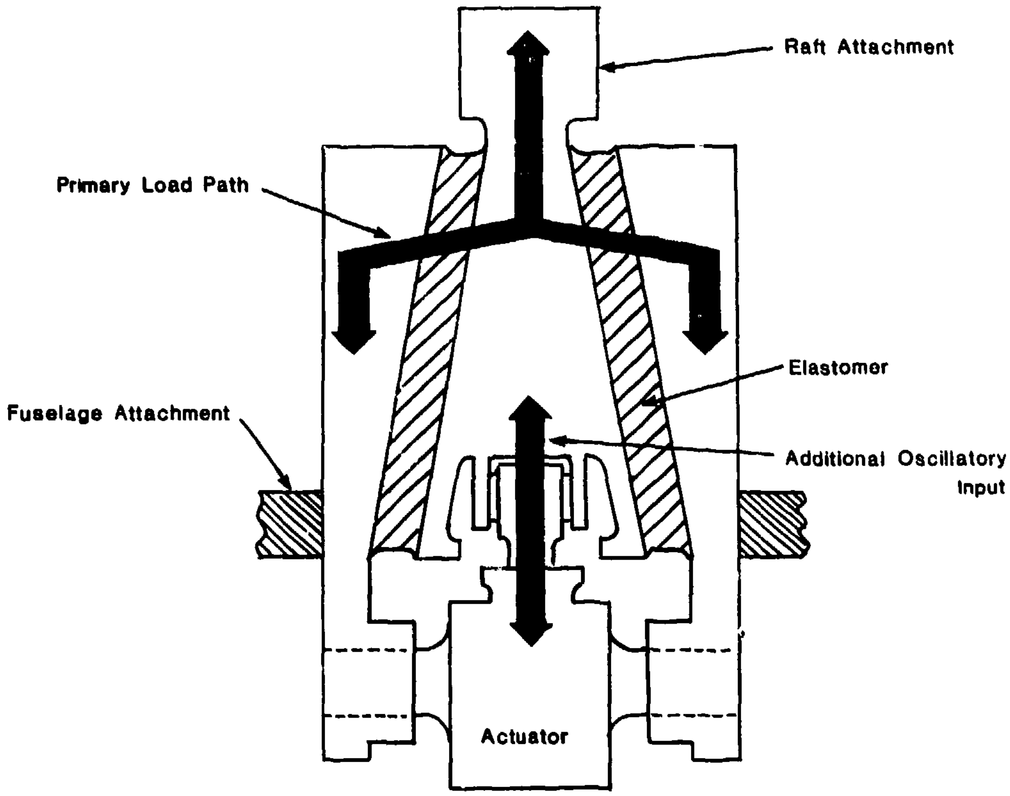


FIGURE 8. Modified Elastomeric Support (Including Force Actuator)

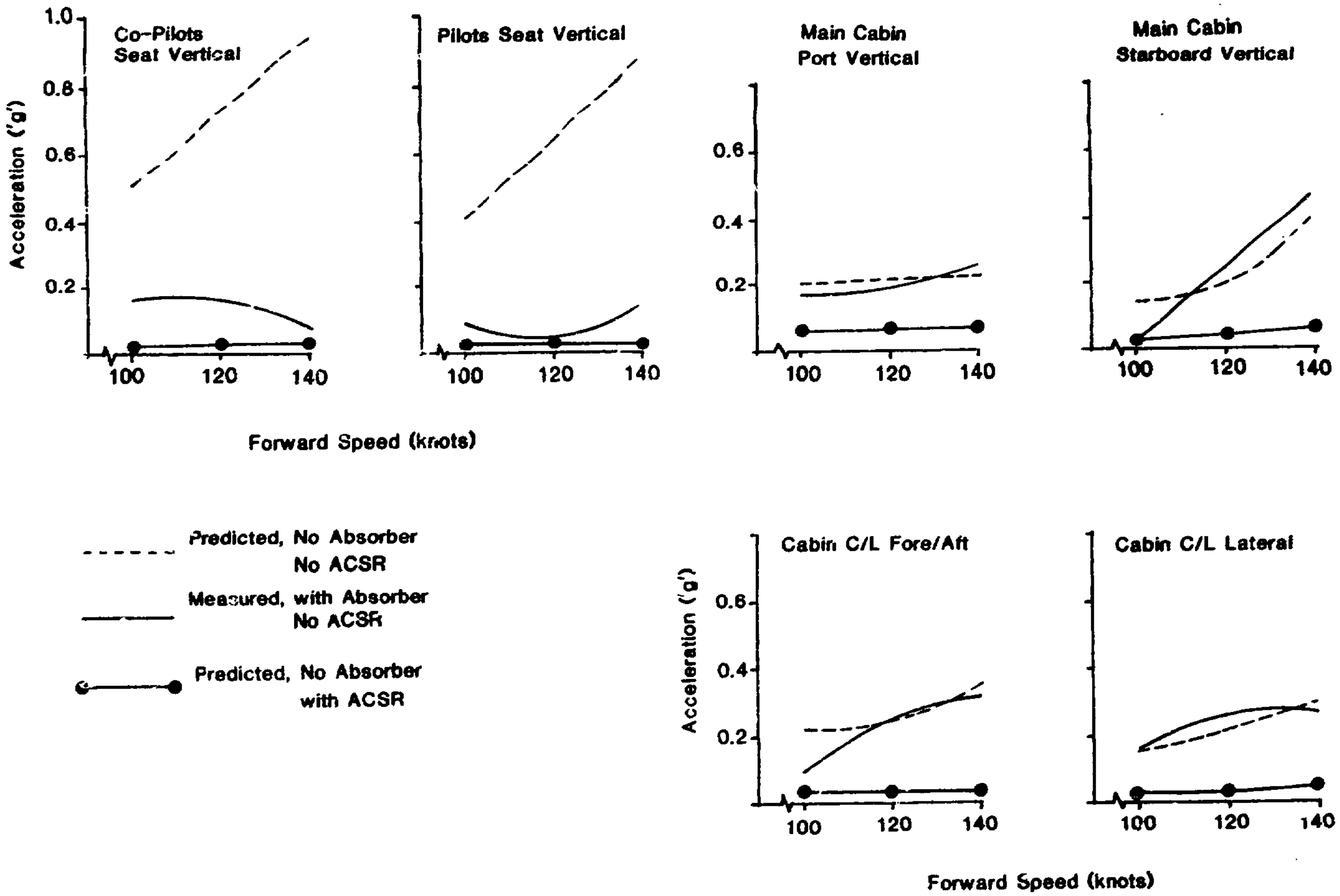


FIGURE 9. Westland 30 Series 100 — 4R Cabin Vibration with ACSR

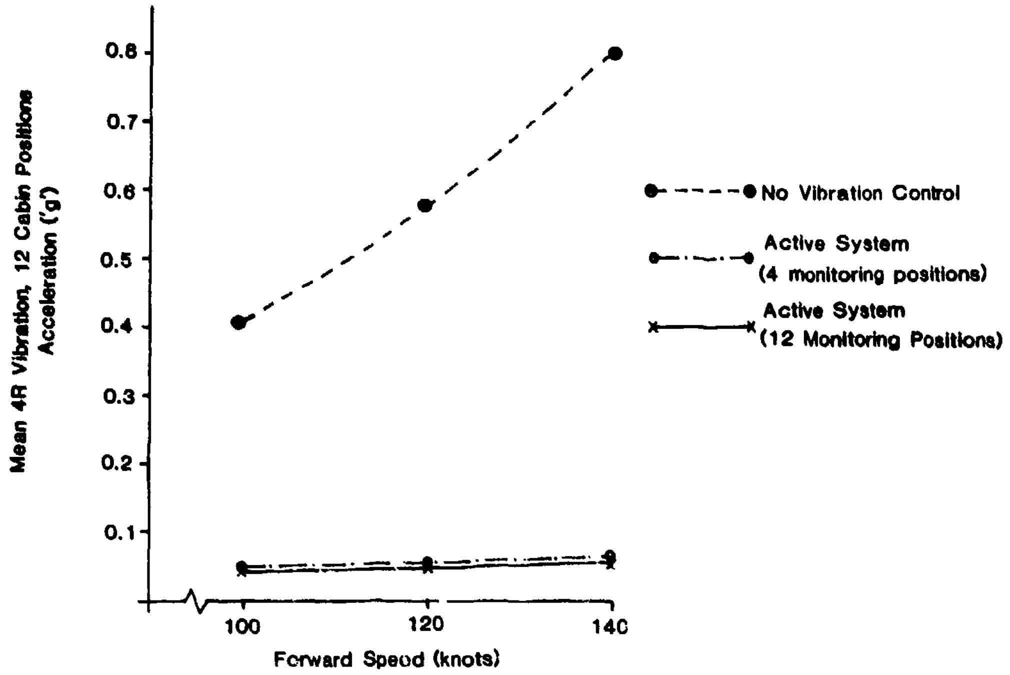


FIGURE 10. Westland 30 (series 100) Predicted Fuselage Vibration

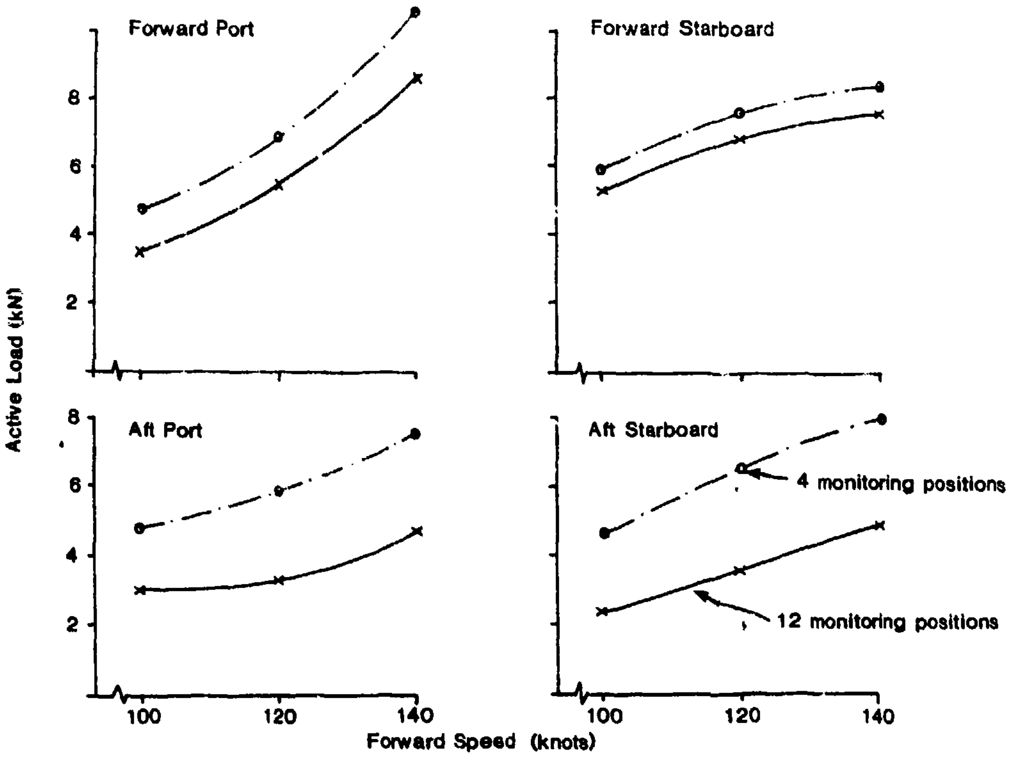


FIGURE 11. ACSR Actuator Loads

IMPACTS OF ROTOR HUB DESIGN CRITERIA ON THE OPERATIONAL CAPABILITIES OF ROTORCRAFT SYSTEMS

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Summary

Starting with the description of the different design principles of articulated, see-saw, hingeless and bearingless rotor concepts, a variety of realised constructions is presented and discussed.

The flight mechanic aspects such as maneuverability and handling qualities of these different concepts are explained with respect to their operational capabilities. In particular, differences in the flapping hinge offset are discussed which have significant influence on helicopter controllability. The trend of the flapping hinge offset over the last three decades of armed and utility helicopters is analysed.

Supplementary to this the improvements in maintainability and reliability are discussed by the increasing simplification of the recent rotor hub designs obtained by the reduction in weight and the lower number of parts.

1. Introduction

Of all the mechanical components of a helicopter, the main rotor head is one of the parts most exposed to stress, caused mainly by the fact that it has to undertake so many functions.
The main function of a rotor head is, among others, to transfer thrust and rotor moments to the fuselage. The high moments which occur at the blade root could not be controlled on the first rotors, which were built similar to propellers, i.e. rigidly attached blades. It was only after the introduction of socalled blade hinges that this problem could be solved. The development went therefore from the simple to the complicated, but only this complicated design made helicopter flight in the past possible. Since then, this type of construction has been used with great success.

It was, however, the desire of all helicopter designers to get away from the complexity of rotor heads, as they were the cause of high production and maintenance costs. However the way back to a simple construction became feasible only when the blade hinges could be replaced by flexible elements with the help of new materials which were able to absorb the high forces by means of elastic deformation. Since that time, a number of different rotor head constructions have met with greater or lesser success.

However, with growing understanding of rotor dynamics, a start was made in using or designing the special dynamic characteristics of the different constructions with respect to the actual mission tasks. In the following, these special characteristics and their effects on the operational capabilities of helicopters are explained in more detail. In the flight mechanics area these factors are maneuverability and flying qualities, in the maintenance sector, the influence on maintainability and reliability. The influence on vulnerability is also shown briefly.

2. Principal Effects of Rotor Heads and Actual Types Produced

In order to be able to show the influence of rotor head designs on the operational capability of a helicopter, the basic structure of various rotor heads should be demonstrated first. Helicopter rotors are usually classified according to the mechanical arrangement of the hub or the attachment of the blades to the hub respectively.

Due to the unsymmetry of the flow through the rotor during forward flight, strong alternating bending loads occur at the blade root for blades rigidly attached to the rotor head. This problem could only be overcome in the past by introducing a horizontal hinge which enables the blade to flap up and down freely (called flapping hinge) and a vertical hinge which enables the blade to carry out movements in the plane of the rotation (called lead-lag or drag hinge). With the feathering hinge about a third axis, usually parallel to the blade span, to enable the blade pitch angle to be changed, the blade is therefore attached with three hinges to the rotor head. This leads to a highly complex structure. Such a rotor head is called fully articulated (see fig. 1).

As an example of the practical design of this is the rotor head of the Sikorsky S-58 helicopter, which is shown in figure 2. As can be seen this rotor head is very complicated.

A first simplification can be seen in the socalled semi-rigid blade attachment for two bladed rotors. Here the two blades are connected rigidly to one another, but they can flap around one common flapping hinge (half gimbal suspension), i.e. when the one rotor blade goes upwards the other goes downwards. This principle is called see-saw rotor or teetering rotor, see figure 1. There are no lead-lag hinges in this design, this task is done by torsional flexibility of the drive system (rotor shaft) or in other designs by 'underslinging' the rotor, so the Coriolis forces can be greatly reduced. Theoretically therefore six hinges are needed in this system for two rotor blades to constitute a fully articulated rotor. This number is reduced now by half. This type of rotor is used mainly by Bell and Hiller. As an example of this type the rotor head of the Bell 206 is shown in figure 3.

The success of the design of the hinged rotary wing has certainly had an inhibiting effect on the development of helicopters and of the aeromechanic and aerodynamic knowlege of the helicopters as well. The hinged rotary wing has a relatively low blade stress, transfers relatively low vibration, has good stability and control characteristics and is relatively insensitive to gusts. These good characteristics connected with a lack of deeper knowledge about the complicated flow conditions at the rotor and the complicated dynamics of this explain the conservative attitude of the successful helicopter firms. For decades the same design principles were adhered to and new models were developed from the empirical experience gained from earlier models. This has changed in the last two or three decades brought forward above all by the general dissatisfaction of the helicopter users with the high maintenance costs caused by the numerous rotor hub bearings which are placed under enormous stress and which have to be greased and to be protected from dust.

Everywhere the rolling or sliding motion is avoided if at all possible and is replaced by flexibility. This has led, amongst others, to a hingeless rotor.

In the case of a hingeless rotor the flapping hinges and lead-lag hinges are removed. Their tasks are now taken over by the elastic deflection of a component (fig. 1). The socalled hingeless rotor concept is not really hingeless, since only the conventional flapping and lead-lag hinges are dispensed with, but not the feathering hinges for pitch control. Truly hingeless rotors without feathering hinges are called bearingless rotors. The hingeless rotor principle was previously also less accurately termed rigid or semi-rigid, although it is in fact not rigid but flexible.

This design was made possible by the introduction of composite materials such as fiber glass reinforced plastic (FGR) which has enough fatigue strength given corresponding elasticity. The omission of the hinges leads to a mechanical simplification and to a better aerodynamic shape of the rotor head, since the rotorhead drag represents a large part of the total drag of the helicopter. Also the maintainability is significantly better as now the feathering hinge is the only rotating bearing present.

Lockheed was the first company to initiate this trend and to carry out successful tests (first flight of the Lockheed Cheyenne was in September 1967, see fig. 4).

The individual helicopter companies have now taken this step back to simplification in a variety of ways.

Lockheed first used steel then titan to make the arms of the rotor hub so flexible that they could allow the flapping motion. The lag motion is thereby completely suppressed. The blade pitch is achieved by socalled door hinges.
This system however was not pursued by Lockheed any further.

Bölkow (today MBB) was then the actual pioneer, with Aerospatiale, of hingeless rotors by the introduction of rotorblades manufactured from glass fiber reinforced plastic (FRP) material. Here the flapping and lagging motions are possible by the deflection of the blade root, whereas the rotorhub itself is very stiff. The blade pitch occurs by a needle bearing whereby the centrifugal forces from the socalled soft torsion bars are transferred to the rotor head (first flight of the BO 105 was in November 1967, see fig. 5).

The Westland Company has so shaped the arms of its titan rotorhead that the inner part withthe elliptical cross section allows the flapping motion by bending this part, whereas the outer arm with the round cross section allows the lag motion. The blade pitch occurs as in the case of MBB via needle bearings, also the centrifugal forces are transferred by soft flexible straps made of steel. However, this rotorhead requires drag dampers (fig. 6, first flight March 1971).
These are the only two hingeless rotors which are currently used on production helicopters, namely on the BO 105 and BK 117 by MBB and on the Lynx and W 30 respectively by Westland.

Other manufacturers followed this development only much later and did not then try to produce a hingeless rotor but instead have tried mainly to replace the mechanical hinges by using elastomeric bearing technology and have therefore remained with the principal of the hinged blades. The elastomeric bearings are made of shaped layers of a rubber-like elastomeric material, so that these bearings can deflect to allow flapping, lead-lag and feathering motions. They are also used to absorb the centrifugal forces . This has been made possible because the flapping and lag motions and also the blade pitch change of the rotor blades only occur in a relatively small angular range.

Considering the history of the development of rotor systems with elastomeric bearings two different ways can be dicerned.

There was on the one hand the introduction of layered elastomeric bearings where an angular motion occured by shear deformation of laminated bearings which allow all three motions. As an example of an articulated rotor using this technology the rotor head of the Sikorsky UH-60 is shown, which first flew in 1974 (see fig. 7). Here all three mechanical hinges have been replaced with a pair of elastomeric bearings which allow the flapping, lagging and torsion motions and also the transfer of the centrifugal forces.
On the other hand, special shaping of the inner part of the rotor hub so that the flapping motion is possible by elastic bending, whereas the lag motion is done by laminated bearings constitutes the other solution. As an example of this the titan rotor head by Bell is shown (see fig. 8) for the model 412 (first flight 1979). The flexible rotor head here allows the flapping motion whereas the elastomeric bearings take on the lag and torsion motion and the lead lag damping.

Rotor design by the former Hughes company (now Mc Donnel Douglas Helicopters) back in 1963 went its own way with its rotor head of the model 500 and then later with the rotor head of the AH-64. Here two opposite blades of the four bladed rotor are connected to the elastic straps (made of steel) whereby blade pitch change, flapping and lag hinges can be dispensed with (see fig. 9).
Aerospatiale was the first company to be successful in replacing the metal rotor head by fiber reinforced plastic material (FRP) with its rotorsystem Starflex in the year 1976 (see fig. 10). The flexible star of the rotorhub made of Composite is designed to accommodate the flapping movement. The blade pitch change and the

centrifugal forces are taken up by a spherical elastomeric bearing. Elastomerics are also provided for drag damping (Ref. 1, 2).
Further developments in this direction are the Spheriflex rotor head for the Super Puma (fig. 11, Ref. 8), the rotor head of the EH 101 (fig. 12) and the rotor head of the AHIP-program of the Bell company (fig. 13, Ref. 26).

A further simplification of the hingeless construction principle is being tried by MBB in the German/French joint project PAH-2/HAP/HAC whereby the very stiff rotor head is being retained, but the titan hub is replaced by two starshaped plates made of fiber composite material. The oil lubricated bearings for the blade pitch control are replaced here by elastomeric bearings (fig. 14).
Due to the relatively short lifetime and the high costs in the initial stages of the elastomeric bearings (this problem has however today been solved to a large extent), the trend of rotorhead development today is quite clearly in the direction of bearingless rotors. A glance at the history of the development of rotors shows this step to be a consistent development trend from fully articulated over the hingeless to the bearingless rotor. Fig. 1 shows the basic structure for this system, the full replacement of mechanical bearings, hinges and dampers leads to an even greater simplification.
The rotor head itself consists only of a very simple hub. In addition to the flapping and lead-lag motion also the torsional motion is done by elastic deflection of a soft torsion element, but relatively stiff flapwise and lagwise (as shown in fig. 1). This element must meet high requirements like low rigidity in torsion, but enough strength to carry the centrifugal forces and the bending loads and above all be designed to specific stiffness in flapwise and chordwise direction. Only the development of composits in recent years has provided the necessary materials for these elements.
A disadvantage of this bearingless design is the relatively long transition path for a torsional moment to the blade root to establish the blade collective and cyclic pitch angle. This pitch control can only be attached outboard of the flex beam. Therefore a torque structure of some kind is required, which is very stiff in torsional direction but soft in bending. The torque structure may either be designed to enclose the flex beam or to be separate from it.
Various design approaches of different companies have been established. They differ in cross section of the flexible element and the stiffness of this element and in the torque structure to carry the torsion loads.

The first successful bearingless rotor systems emerged from the UTTAS competition in the USA. Both Sikorsky (Ref. 3) and Boeing Vertol (Ref. 4) used stiff inplane (i.e. blade inplane frequency ω_{γ} is above one per rev.) bearingless designs for their tail rotors. The introduction of this design principle for tail rotors appeared to be considerably simpler as shown on many similar prototypes e.g. at MBB (Ref. 5) and Hughes (Ref. 6).
The feasibility of this construction principle of main rotors was proved for the first time by Boeing Vertol with their Bearingless Main Rotor (BMR) on a BO 105 (Ref. 7). This rotor represented a good basis for future development in this area.
A problem with bearingless rotors is the transfer of control movements to the blades as mentioned before. With the BO 105/BMR which undertook its first flight in 1978, this was done by using a torque tube (fig. 15) that does not enclose the flex beam. Other manufacturers have solved this with a socalled pitch cuff that encloses the flex beam.

Whilst Boeing Vertol developed the BMR, Aerospatiale worked on its version of the BMR, the socalled Triflex. The hub consists of a number of fiber glass/epoxy resin rovings which are embedded in a matrix made of elastomer material and which forms a flexible arm. The drag damping of this system was however so low that the system was not pursued any further (fig. 16, Ref. 9).

The bearingless Bell 680 main rotor (fig. 17) flew for the first time in May 1982. It consists of a single fiberglass structure which forms the flexible arms for all four blades. The blade pitch control is transferred by a pitch cuff which surrounds the flexible arms. The part of the cuff on the side of the rotor head is attached by an elastomeric bearing and connected to a drag damper (Ref. 10).

MBB flew its first experimental BMR in 1984. The way it worked was similar to that of the BMR by Boeing Vertol with a T-shaped flex beam and a control torque tube beside the flex beam. The second BMR version with a pitch cuff flew for the first time in 1986. A lead-lag damper between the cuff and the blade root is installed to avoid instabilities in flight and on the ground (fig. 18, Ref. 11).

The Hughes company has also developed a bearingless rotor (HARP). It flew for the first time in April 1985 (fig. 19). It consists of four single flexible arms which are made of Kevlar. Here the cross section of the flex beam is X-shaped. A cuff is used for pitch control. The cuff attachment and the dragdamper are similar to those of the Bell 680 rotor (Ref. 12).

Sikorsky's project for a bearingless main rotor is the socalled Dynaflex, which, however, has yet to be built and flown. The rotor hub (fig. 20) has an elastomeric connection with the rotorshaft which corresponds to a gimbal hinge which permits the flapping motion and, by preventing the Coriolis forces, prevents the lead-lag motion. The torque transmission and the flapping spring are obtained by means of a rotor head surrounding membrane in composite material. This rotor concept has therefore the lowest aerodynamic hub drag of all BMR designs up to date. A central torque tube is provided for pitch control similar to that of the BMR of Boeing Vertol (Ref. 13).

All bearingless rotor designs flown up to now were designed for light helicopters of an all-up weight up to 4 metric tons. For larger helicopters this design principle has not been realized yet.

3. Principle Differences of the Various Rotor Head Systems with Reference to Controllability and Flying Qualities

In this section I want to treat the special flight mechanical characteristics which these various rotor systems have. The dynamic behaviour of a helicopter depends to a large extent on the moment that is transferred from the blades to the rotor hub and to the fuselage. There are fundamental differences here between the various rotor systems and these are dependent on the flapping hinge offset. Fig. 21 shows qualitatively the moment characteristics (first harmonic) of a see-saw rotor, an articulated and a hingeless rotor. If the aerodynamic lift of the blade is the same for the various rotors, the same lift is produced at the centre of the rotor head. But for the moments there is a significant difference at the blade root and at the head. A hinge cannot transfer moments, so there is no moment in a see-saw rotor head because of the central flapping hinge on the rotor head. Articulated rotors have in general a small flapping hinge offset whereby only small moments can be transferred as a result of the short moment arm. Hingeless rotors have a special area of high elasticity either in the blade or in the head itself with relatively low moments which can also be described as equivalent flapping hinge. The offset of this equivalent flapping hinge depends on the stiffness of the flexible element. Usually this 'hinge' is further away from the rotor centre and can therefore transfer larger moments because of the longer moment arm.
Hingeless rotor systems and bearingless rotor systems are best characterized by the fundamental flap (ω_β) and lead-lag (ω_ζ) natural frequencies of the blades at nominal rotor speed (Ω). The see-saw rotor type with its central hinge (without spring restraints) would have a natural flap frequency ω_β equal to 1Ω. Articulated rotorsystems with flapping hinge offset usually have a flap frequency greater 1Ω to about 1.04Ω.
For hingeless and bearingless main rotor configuration the range of the fundamental flap bending frequency ω_β lies between 1.05Ω and 1.15Ω. From this fundamental flap bending mode an equivalent flapping hinge offset can also be derived.

The control of a helicopter occurs mainly by the tilting of the thrust vector and the moment which is produced at the rotor head. From that it follows that in the case of the see-saw rotor the control only occurs through the tilting of the thrust vector which only permits a restricted control moment. Bell tried to overcome this problem on some of their types by incorporating a spring in the teeter hinge.
The articulated rotor has increased control moment capacity according to the flap-

ping hinge offset, whereas the hingeless rotor has greatest control power due to its relatively large flapping hinge offset, whereby by far the greatest proportion is alloted to the moment at the rotor head. In figure 22 this situation is presented schematically.
The direct consequence is a reduction of the helicopter time constant. This time constant is defined as the time which is needed to achieve 63% of the stationary angular velocity at a defined control input. The time constant is shown in figure 23 and 24 as a parameter for pitching and rolling with various flapping hinge offsets. As can be seen the control of a helicopter with increasing flapping hinge offset becomes more powerful, quicker and more direct, that means it achieves higher control response characteristics.
It should be mentioned that the achievable steady-state angular rate is independent of the flapping hinge offset (the flapping hinge offset influences the transient phase only).

These characteristics however make the rotor with large flapping hinge offsets also sensitive to disturbances from outside such as for instance gusts which can be considered as control inputs.
The control sensitivity of a system depends however also on the rotor damping. By this is meant the increasing damping moment during the pitch or roll motion that the motion reacts against and which leads to a steady-state angular rate at a specified unitary stick deflection.
To judge the control characteristics the control and damping moments at the rotor have to be set in conjunction with the moments of inertia of the helicopter. The well known controllability diagram (fig 25) relates the control sensitivity and the time behaviour of a rotor to one another.
This picture shows the damping moment versus the control moment produced per unit stick deflection for the pitching direction. Both values refer to the moment of inertia around the pitch axis. For a favourable assessment of the control characteristics of a helicopter these values must be in a particular relationship to one another. Along the straight lines through the origin there is equal control sensibility i.e. equal angular velocity -in this case pitch velocity- per unit stick deflection. In this illustration recommendations or requirements respectively are included (Ref. 14, 15, 27, 28). The range of them is established by certain minimum and maximum pitch speeds for a unit stick deflection and downwards by a minimum damping. As the control sensitivity can be changed by the selection of the control ratio in a certain area, the non influencable value of the rotor damping represents the decisive value for position in the diagram. Here the typical areas for the different rotor systems are included. For the roll axis there is a similar correlation.

The damping and the control acceleration depend generally on the Lock number - that is the number which represents the ratio of the aerodynamic to the inertial forces of a rotor blade - and the flapping hinge offset. In comparison to the hingeless rotor system helicopters with articulated or see-saw rotors have considerably less control response characteristics as a result of the high time constant.

The control power of the various systems - on the one hand through the thrust vector tilt only and on the other through the thrust vector tilt and rotormoment - have decisive influence on the flight conditions with thrust off loading, that is a flight condition in which the load factor is smaller than one or even close to zero. These flight conditions occur for instance when overflying obstacles which appear suddenly during terrain following flight (NOE). Figure 26 shows the dependence of the control moment on the load factor for the three rotor systems. Whereas the controllability of the systems with high flapping hinge offset basically remains the same, the see-saw rotor shows a percentual decline in its controllability at n less than 1 and leads finally at n = 0 to incontrollability.

As mentioned briefly above adverse effects also go along with increasing rotor head moments such as higher vibrations and gust sensitivity and higher angle of attack instability.
To look at other flight mechanic aspects such as high speed flight instability phenomena or effects of different couplings which can mainly be influenced on hingeless and bearingless rotors and other aeroelastic problems would be to go beyond the limits of this lecture.

However it can be said quite generally that special mission tasks can be more or less well solved with the different rotor systems. The idea of optimising a system for its specific mission tasks suggests itself. A rotor design of course depends on many things which have to be coordinated to an optimum degree for one another but one can say that perhaps combat helicopters or scout helicopters should be equipped with a rotor system which fulfils the special tasks of these helicopters well such as for instance nap of the earth (NOE) flying which presupposes high maneuverability. On the other hand this high maneuverability is not so decisive in the case of -for instance- transport helicopters for them with regard to their particular mission.

The history of rotor development shows that the individual helicopter manufacturers have developed their own philosophies with reference to rotor systems and to their design. Manufacturers of stiff rotors, which are therefore those with relatively large flapping hinge offsets (i.e. hingeless rotors), have the view that this method of design is best adapted to future combat helicopters. However, as has been shown this method of construction contains characteristics through higher vibration and gust sensitivity which are not desirable for those helicopters which are deployed as weapon platforms. Therefore in recent times the greatest efforts have been made to avoid these problems by active and passive vibration isolation systems. All together, however, one is in a dilemma as to these two demands on a combat helicopter - mainly on the one hand the greatest possible agility achieved by high excess power and high control response characteristics or maneuverability and on the other hand low vibration and low gust sensitivity - presuppose completely contrary design principles. Other manufacturers of 'softer' rotors (i.e. articulated rotors) also believe that this design is well adapted to the requirements of combat helicopters through better vibration characteristics and less sensitivity to disturbances from outside.
Only the hingeless and the bearingless rotor concepts are very sensitive to small parameter variations, but on the other hand these rotor concepts can be designed through proper aeroelastic couplings to suit these requirements (Ref. 25) extremely well.

Figure 27 shows the temporal development of rotor rigidity of armed helicopters expressed by means of the flapping hinge offset. Although it is difficult to establish a trend, it can be stated that the future development will probably settle at a flapping hinge offset of about 7% to 8% which corresponds to a blade flap bending frequency of about 1.06Ω. Therefore in this type of helicopter the see-saw rotor will no longer be used. Future helicopters in this weight class have therefore a well tuned and aeroelastically balanced bearingless rotor with an equivalent flapping hinge offset of about 7% or 8%.

In the transport helicopter class in which a bearingless design has not yet been realized, future developments will still have articulated rotor designs but with extensive use of laminated bearings. The trend in the development of rotor rigidity for these helicopters is shown in figure 28. Here the developments will probably remain more conservative as no high maneuverability is demanded for such systems. The priorities of the development will be aimed in other directions such as for example transport performance but also passenger and crew comfort. Here in future a flapping hinge offset of 4% to 5% will be the norm. This is also influenced by the difficulty of achieving such a 'soft' rotor with the bearingless design.

4. Influence of the Different Rotor Systems on Weight, Maintenance and Reliability

In addition to the design of the rigidity of the rotor, a further factor lies in the actual construction which determines the number of parts and the maintainability with all its consequences and this is of significance for the operational capabilities of a helicopter. As mentioned before, it was both the expensive maintenance and repair actions on the rotor head as well as the increased military requirements against vulnerability which were the driving forces for modern developments. Fully articulated rotor systems which were until now still very widespread had many grease points and were, due to the many parts made of steel, very heavy. The trend for future rotors can be seen in the next figure (29). The proportion of the rotor head mass was in the articulated rotor of the older generation

between 6% and 7% of the maximum take off mass, which was reduced in the 1970's to about 3% to 4% by the introduction of the elastomeric bearings. See-saw rotors were naturally lighter (because of only two blades), but they still had a relatively high proportion of the maximum take off mass of about 5%. The introduction of hingeless rotors brought a reduction of the rotor head masses by the omission of the flapping and drag hinges. This can mean for future bearingless rotors a percentage of 2% or less.

In addition to the payload increase there is the decisive advantage for future rotor developments in the service and maintenance actions which are directly proportional to the construction concept and connected with a reduction of the aircraft support costs.
A large part of the maintenance of helicopters is devoted to the two rotors (the main and the tail rotor) and the rotor blades.
By the introduction of fiber reinforced plastic (FRP) rotor blades by MBB the first important steps were taken with relation to reduction in maintenance costs, life time and vulnerability. The practically world wide application of this material has proved the outstanding characteristics of fiber glass above all the inherent fail-safe behaviour, the very good resistance to fatigue and the complete insensitivity to notch and corrosion effects.
The major progress made in blade aerodynamics can be mentioned here only in passing. Complex blade forms, variable profiles over the blade length and twist and new blade shapes can only be made with the help of composite material without increasing the manufacturing costs (Ref. 19, 20, 21, 22, 23, 24).
Special tests have also proved that the composite rotor blades and also of course composite rotor heads are distinguished by very high survivability in a combat surrounding (see fig. 30 from Ref. 17). Today there is practically no helicopter manufacturer in the western world who does not at least for its future helicopter wish to use FRP, for the rotor blades.

The reduction in the number of bearings from an articulated rotor to a hingeless rotor brought a reduction of maintenance efforts -if the two systems are compared- of a factor of 3. This is also because the hingeless rotor has up to 65% fewer parts.
These points are noticable in the availability but very particulary in the field maintenance. The logistic requirements are therefore considerably reduced.
But the high requirements of the military from scheduled maintenance to 'on condition' maintenance could only be achieved first with the introduction of the elastomeric bearings which do not have to be lubricated. The relatively short life time of these elastomeric parts in the beginning have now reached a high technical standard (life time ca. 2500 hours). The successful use of FRP rotor blades has in the course of development of new rotor head constructions led to the fact that this material is also used for rotor heads, whereby the number of parts has again been considerably reduced (Ref. 1, 18).
Future bearingless rotor heads will have a parts counts of 20% as compared to fully articulated rotors due to the complete absence of bearings (see fig. 29). The availability - expressed in meantime between failure (MTBF) and mean time between removal (MTBR) - and the maintainability - expressed in mean manhours per flight hours (MMH/FH) and mean time to repair (MTTR) - will improve significantly (Ref. 16). The use of FRP for future bearingless rotor heads and rotor blades will further reduce the vulnerability, thus an important military requirement will then be fulfilled. This again directly influences the performance of military helicopters in the fulfilling of their mission.

In contrast to the dynamic design of the rotor with different rigidities the requirements on the rotor construction - i.e. low weight, low number of parts, high lifetime, easy maintainability, etc. - both for civil and for military users are basically the same as they lead to increased performance, lower procurement costs and lower service cost. For the military side the high demands on survivability are added.

5. Conclusion

The different needs of the individual users of helicopters and reasons which were dealt in chapter 2 have in the past produced a large number of rotor head configurations. The aim now should be to have a single ideal rotor concept which satisfies all demands. This dream of helicopter engineers will -as we have seen above- not be fulfilled as differing philosophies and different operational needs will not permit this.

The objective in the development of future modern rotor systems with regard to the dynamic design for the specific military employment will as before be directed at

- high maneuverability
- uncritical stability behaviour
- low vibration generation

This will lead to a design where the equivalent flapping hinge offset will settle at about 7%. This presents a good compromise between high maneuverability and not too high gust sensitivity.

The general objective in the development of modern rotor systems and its influence on the operational capability can be summarized as follows

- reduction of the weight by the omission of conventional bearings and hinges

- increase in lifetime by the omission of high maintenance and mechanical elements subject to intensive wear

- increase of safety by simple construction reducing the number of parts and the use of fail safe characteristics of composite materials

- reduction of the maintenance efforts by maintenance free bearings and hinges

- reduction of manufacturing and operating cost by simple construction, a low number of parts and 'on condition' maintenance and unlimited lifetime of essential components.

These high technological goals present a challenge for all rotor head constructors for the next decade.
First steps in this direction have been taken, but it still requires great technical efforts and a high financial investment of all helicopter manufacturers to do justice to all the demands. Bearingless rotors perhaps present the best assumption for this.

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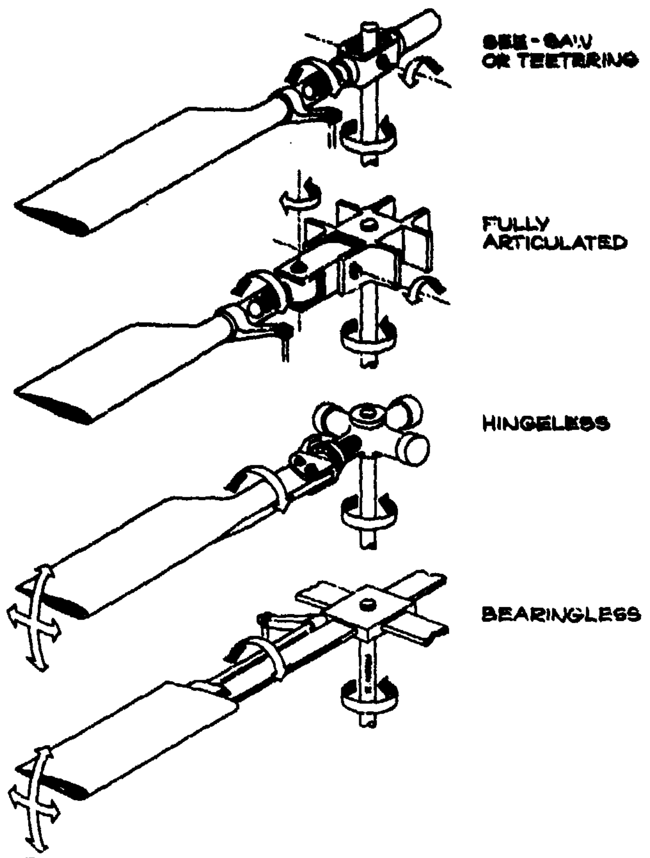

FIG.: 1 PRINCIPLE OF DIFFERENT ROTOR SYSTEMS

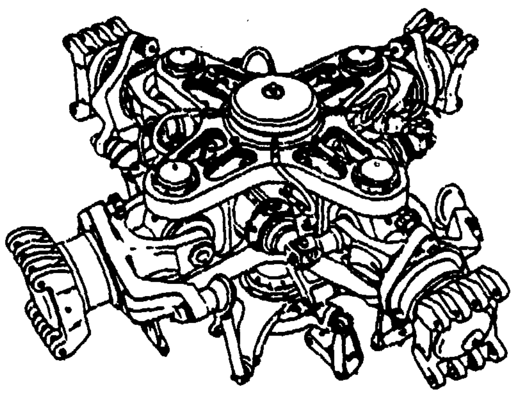

FIG.: 2 ROTOR HEAD OF SIKORSKY S-58

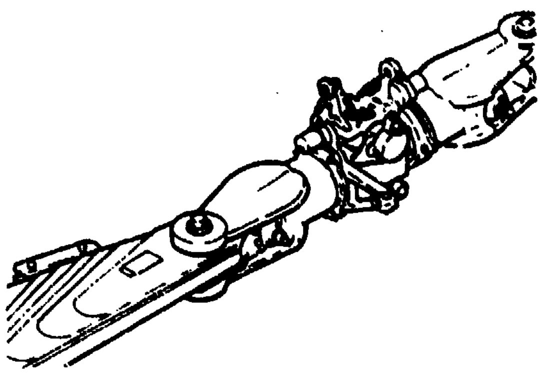

FIG.: 3 ROTOR HEAD OF BELL 206

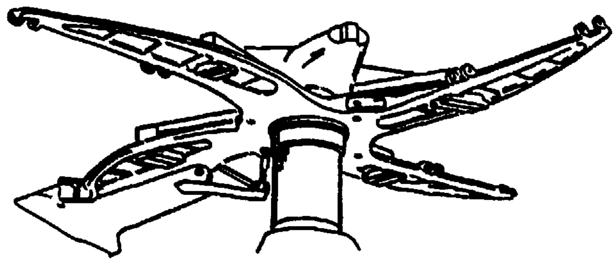

FIG.: 4 ROTOR HEAD OF LOCKHEED CHEYENNE AH-56 A

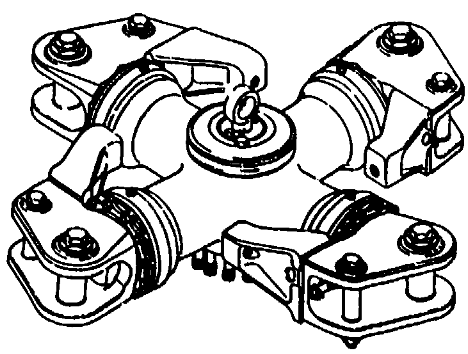

FIG.: 5 ROTOR HEAD OF MBB BO 105

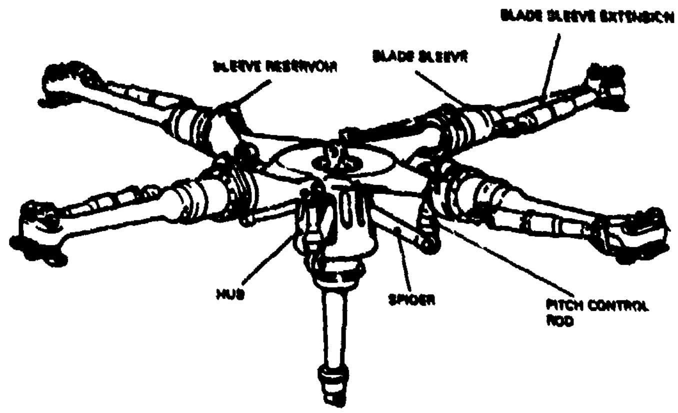

FIG.: 6 ROTOR HEAD OF WESTLAND LYNX

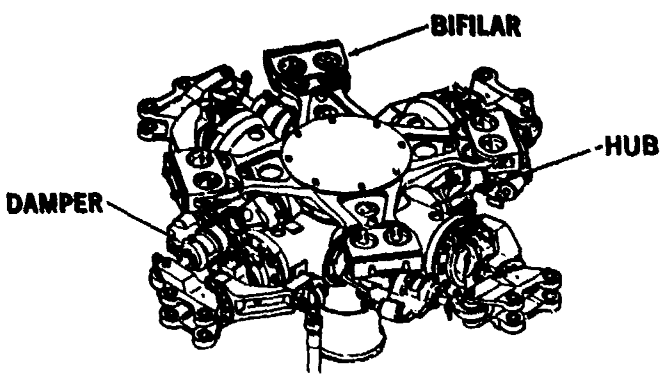

FIG.: 7 ROTOR HEAD OF SIKORSKY UH-60A

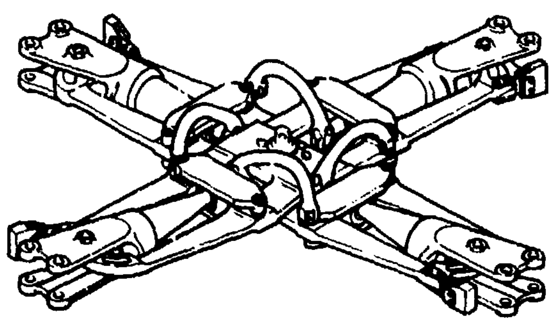

FIG.: 8 TITAN ROTOR HEAD OF BELL 412

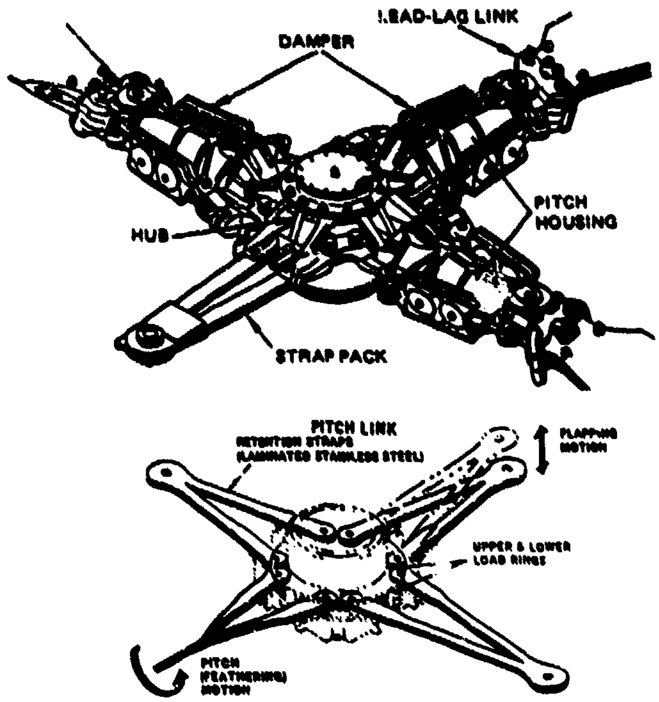

FIG.: 9 ROTOR HEAD OF HUGHES AH-64

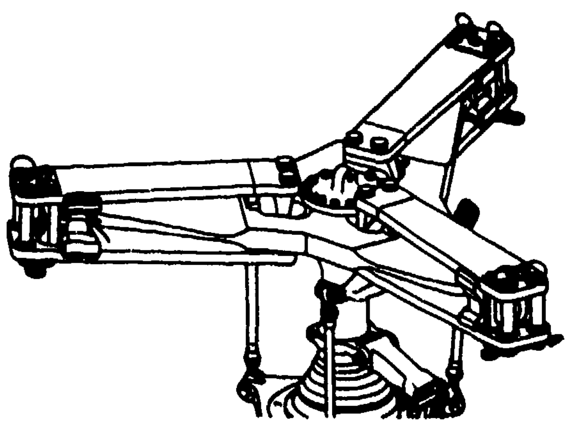

FIG.: 10 STARFLEX ROTOR HEAD OF AEROSPATIALE AS 350/355 ECUREUIL

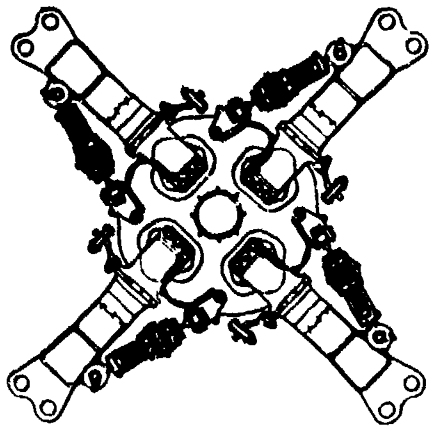

FIG.: 11 SPHERIFLEX ROTOR HEAD OF AEROSPATIALE AS 332

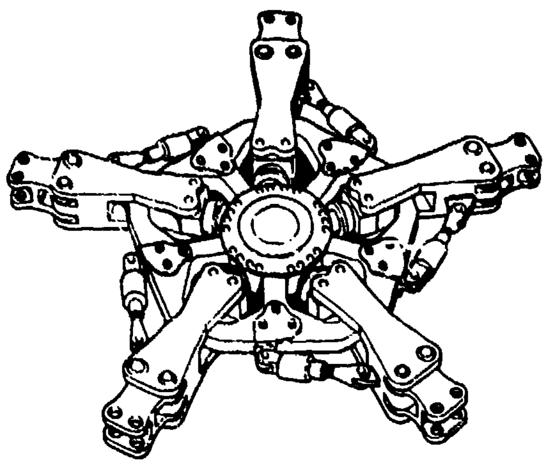

FIG.: 12 ROTOR HEAD OF AGUSTA AND WESTLAND EH-101

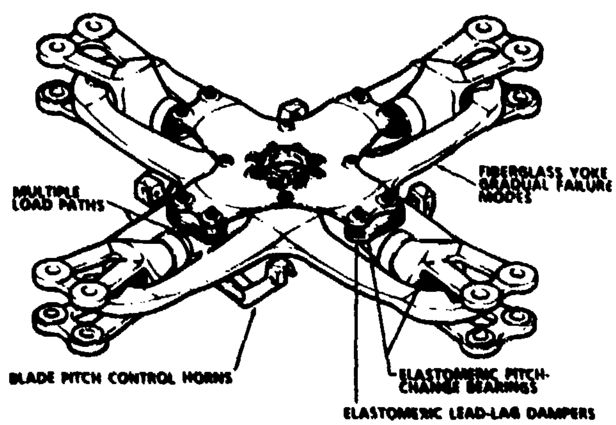

FIG.: 13 ROTOR HEAD OF BELL OH-58D (AHIP)

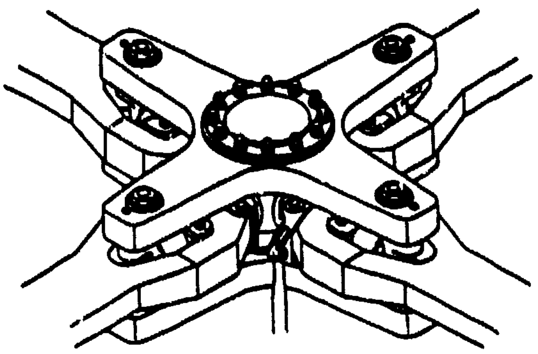

FIG.: 14 FEL ROTOR HEAD OF MBB PAH-2/HAP/HAC

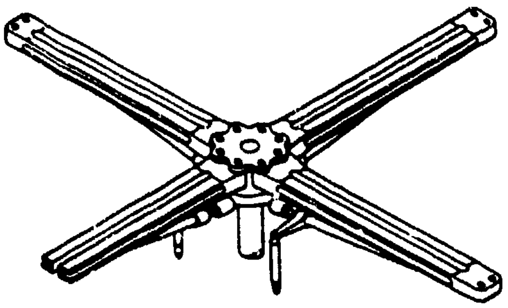

FIG.: 15 BEARINGLESS MAIN ROTOR HEAD OF BOEING VERTOL FOR BO 105

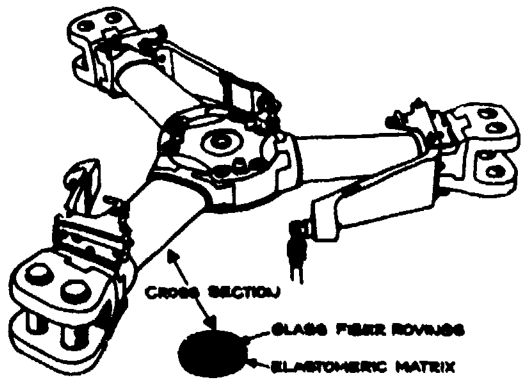

FIG.: 16 TRIFLEX ROTOR HEAD OF AEROSPATIALE

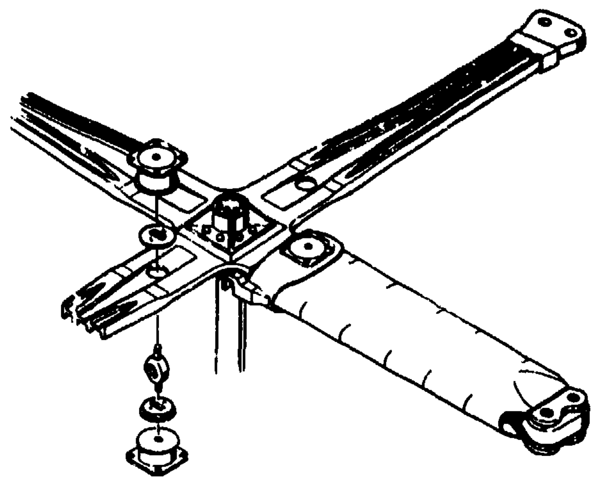

FIG.: 17 BEARINGLESS ROTOR HEAD OF BELL 680

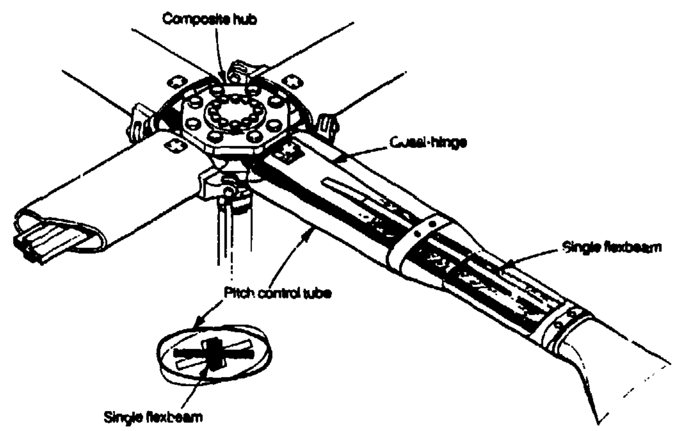

FIG.: 18 BEARINGLESS ROTOR HEAD OF MBB

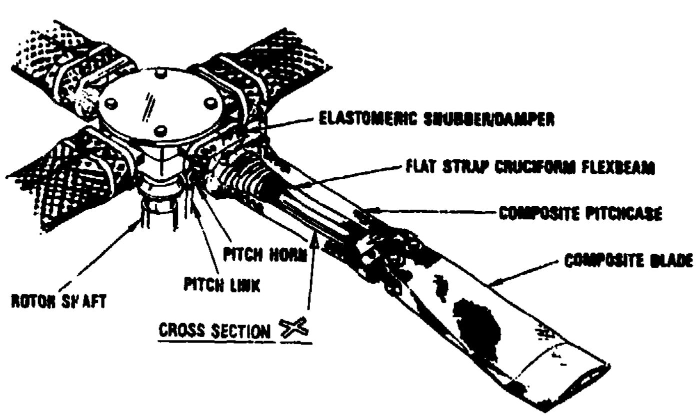

FIG.: 19 BEARINGLESS ROTOR HEAD OF HUGHES HARP

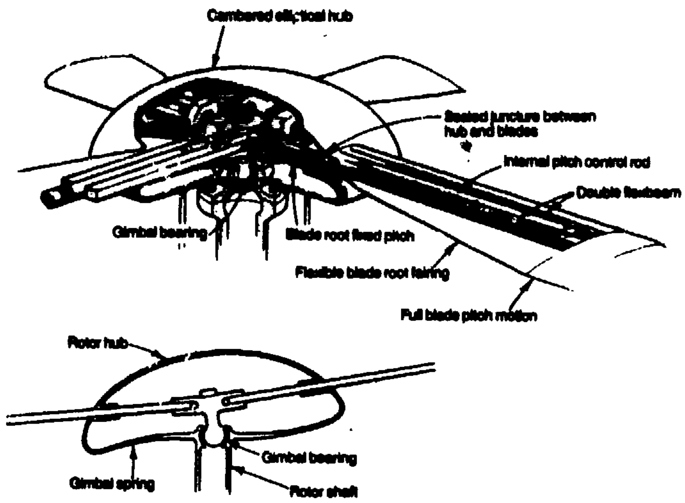

FIG.: 20 DYNAFLEX ROTOR HEAD OF SIKORSKY

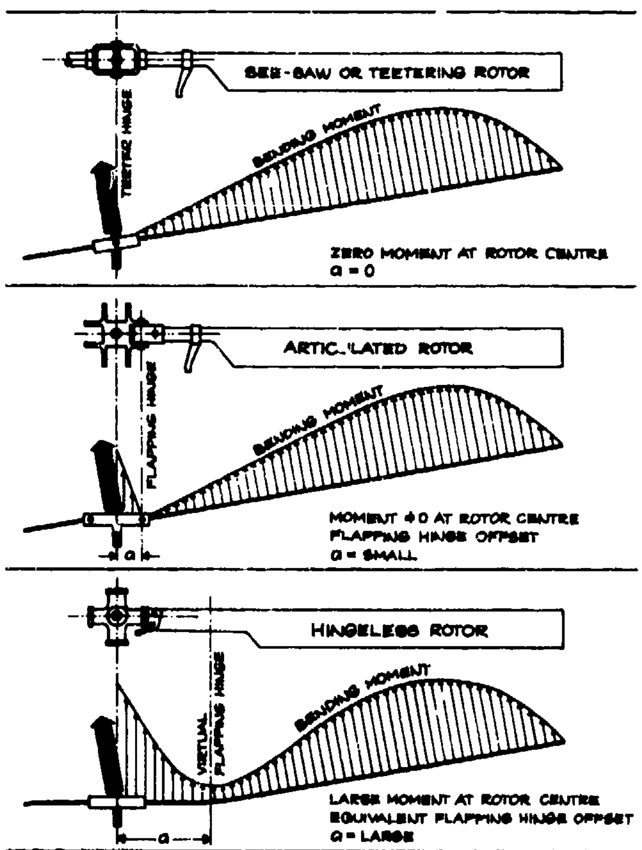

FIG.: 21 BENDING MOMENTS DEPENDING ON THE POSITION OF THE FLAPPING HINGE OFFSET a

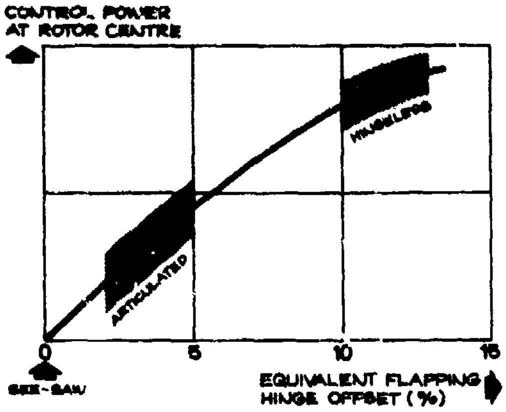

FIG.: 22 CONTROL POWER VERSUS FLAPPING HINGE OFFSET

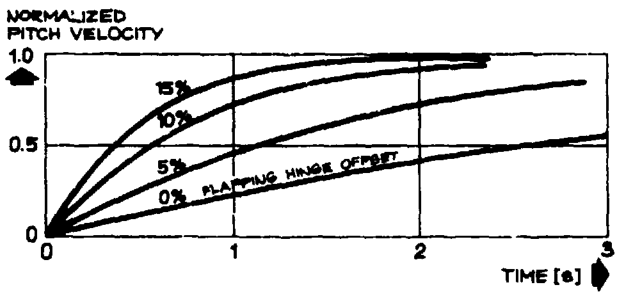

FIG.: 23 TIME HISTORY FOLLOWING A PITCH CONTROL STEP INPUT

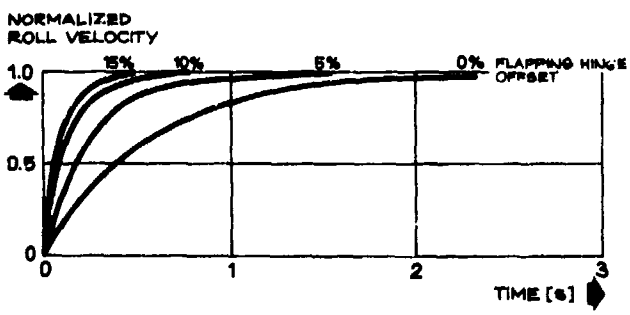

FIG.: 24 TIME HISTORY FOLLOWING A ROLL CONTROL STEP INPUT

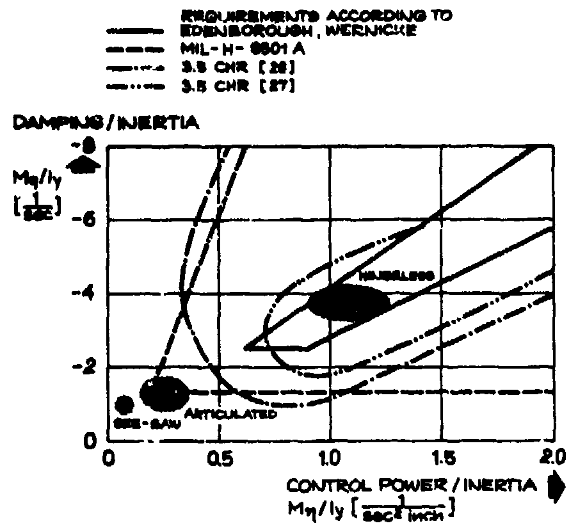

FIG.: 25 HELICOPTER CONTROL RESPONSE CHARACTERISTICS - PITCHING

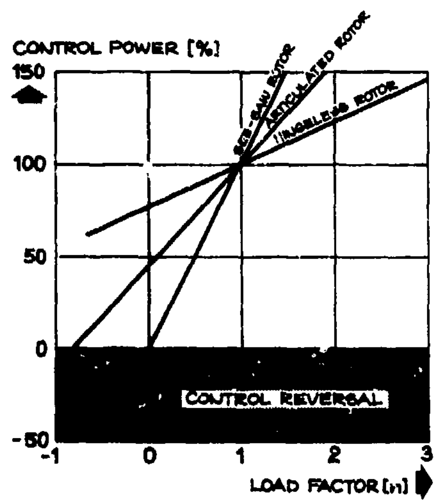

FIG.: 26 MANEUVER CONTROL POWER

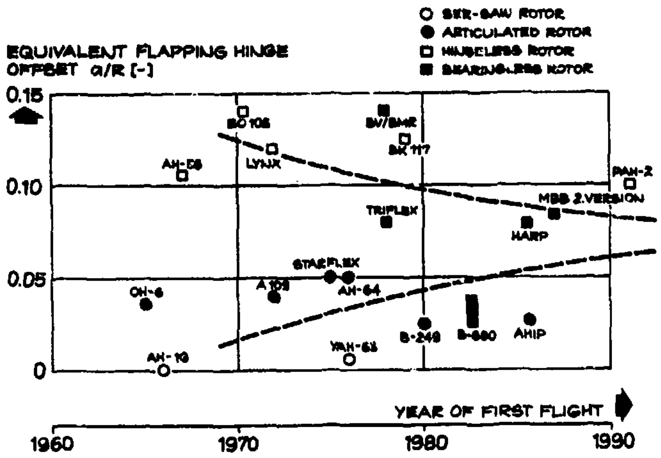

FIG.: 27 TRENDS OF THE FLAPPING HINGE OFFSET OF ARMED HELICOPTERS

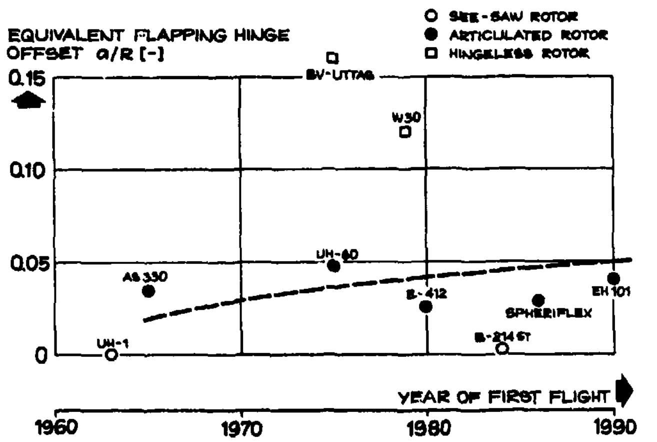

FIG.: 28 TRENDS OF THE FLAPPING HINGE OFFSET OF TRANSPORT HELICOPTERS

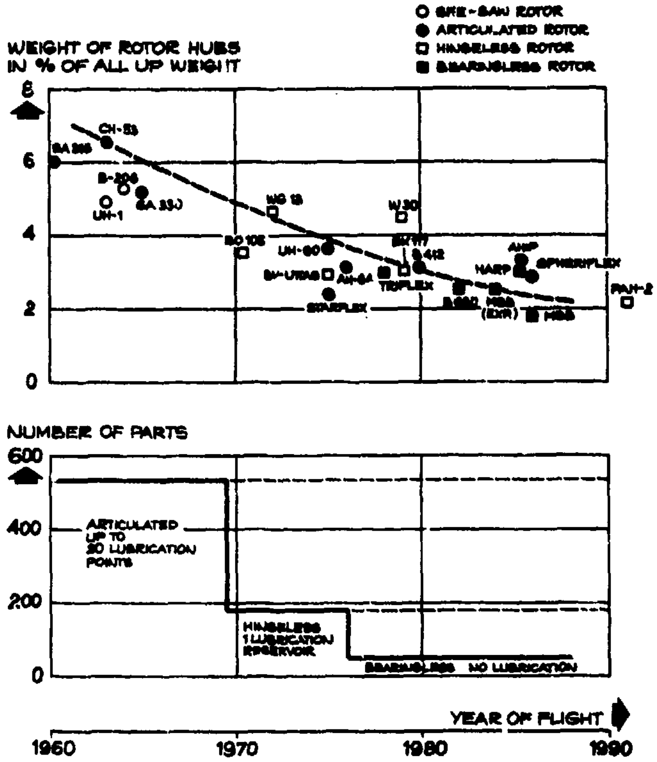

FIG.: 29 NUMBER OF PARTS AND WEIGHT OF ROTOR HUBS VERSUS PRODUCT INTRODUTION TIME

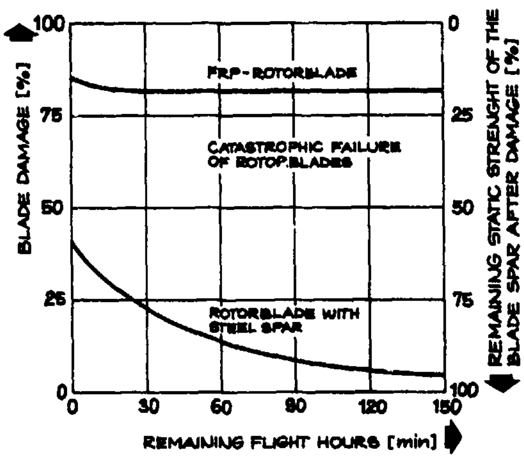

FIG.: 30 REMAINING LIFETIME AFTER DAMAGE OF ROTORBLADES (REF.: 17)

NEW AERODYNAMIC DESIGN OF THE FENESTRON FOR IMPROVED PERFORMANCE

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Aerospatiale PANTHER prototype equipped with the fenestron

ABSTRACT

Since the first Gazelle flight in 1968, Aerospatiale has developed the fenestron as an alternate solution to the conventional tail rotor for light or medium helicopters weighing less than 6 tons. This concept has widely evidenced its advantages on about 1100 Gazelle and 235 Dauphin helicopters equipped with this fenestron and totalizing more than 2 million flight hours, without any major accident. The paper first recalls the general definition of the fenestron and its advantages for civil or military applications.

Recent research has shown new opportunities for improving the aerodynamic efficiency of this fenestron. A detailed airflow analysis through the fenestron has recently been achieved with extensive model and full scale tests on the tail rotor bench in hover. The research program was sponsored by the French Government Agencies DRET and STPA. This paper surveys the experimental technique and the flow measurements. It also presents the correlations that have been made with blade element theory as well as a more advanced analysis developed by METRAFLU and derived from a radial equilibrium code in use for compressors.

The tests have authorized more thorough flow investigations which have shown potential benefits in recuperating the rotational energy. This has led to design stator blades located behind the rotor, inside the diffuser. Tests of this device have shown large improvements in the fenestron's figure of merit and maximum thrust, for a given rotor blade solidity. Furthermore, improving the diffuser's performance, the stator blades permit reducing the diffuser's length and thus the fenestron's width with drag savings as a final result. Specifications were drawn up for ONERA to design a set of specially adapted, high cambered airfoils in view to further increase the maximum thrust.

Tests of the fenestron equipped with stator blades and new sections are presented and their influence on fenestron sizing is discussed.

These various results will further enhance the fenestron performance which has already proven quite advantageous compared to the conventional tail rotor for several decisive points such as safety, reliability, performance and cost for civil applications as well as detectability and vulnerability for military applications.

1.0 INTRODUCTION

The qualities requested for present and future helicopters from an operator view point, are essentially:

- better efficiency
- improved security and reliability
- excellent cost effectiveness

The civil operator will normally be well satisfied if the manufacturer could prove that his helicopter is indeed outstanding on the above qualities. At the utmost, he may also request a high level of availability, but this fourth request is more or less embedded in the previous three.

The military operators have their own special requests depending on the type of missions that they have to fulfill and so they have to accept various types of trade-off. They will at least request low vulnerability and good crashworthiness behaviour.

In this general context, one can ask if it is worth spending time and money to try to develop better tail rotors.

A brief set of data can easily illustrate that the answer is yes:

a) The number of helicopters crashed due to failed or impacted tail rotors is about 0.15 per 10,000 hrs of flight in the accident log book, as compared to a registered overall number of accident of 0.71 per 10,000 hrs of flight.

b) Tail rotor noise can represent a significant part of the helicopter acoustic signature at least in one flight path of the ICAO procedures retained for noise certification: the take-off (see ref.[1]). Furthermore on an acoustic detectability standpoint, conventional tail rotors with high acoustic energy content at low frequencies, can be the dominant noise source at large distances.

c) Tail rotors of improved design can on a given aircraft reduce the power needed for maximum tail rotor thrust, improve the maximum thrust capability and reduce the component weight to thrust ratio. When no other constraints are encountered (available power, gear box limitation, structural strength of the helicopter), tail rotor improvement can allow for an increase of the helicopter payload or of the helicopter flight envelope.

Aerospatiale has studied several tail rotors on various helicopters, ref.[2], and has developed an original tail rotor concept the "fenestron", to overcome the major drawbacks of conventional tail rotors.

On the fenestron, the rotor is housed in a shroud which protects it naturally against most of the agressions, reduces the radiated noise and provides several advantages in operation which will be quickly recalled. This paper will then concentrate on the fenestron aerodynamic development in hover, for which recent research has given new opportunities for improving its performance.

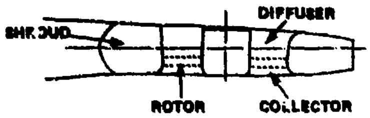

Fig.-1- AS 365 M PANTHER fenestron

2.0 GENERAL DESIGN AND TECHNOLOGY EVOLUTIONS

FIG.1 shows the outline of the AS 365M PANTHER fenestron. The assembly is composed of a small rotor housed in a shroud and topped with a large vertical fin. The rotor diameter is almost one half the equivalent conventional tail rotor diameter and the rotor solidity is roughly twice. So, the blade area is also reduced to one half.
The shroud includes a small collector with rounded lips, a small cylindrical zone at the blade passage and a conical diffuser accommodating the transmission tube, the gearbox with its support arms and the pitch control system.

The first fenestron was flown on a prototype GAZELLE helicopter in April 1968. The production aircraft fenestron had a 700 mm diameter rotor. The blades were made of forged metal and were linked to the hub through a set of thin stainless steel strips of small torsional rigidity to ensure pitch variations. The hub, which holds the self-lubricating plastic type bearings to cantilever the blades, was machined from a light aluminium alloy stamping. The shroud and the fin also are metallic. Ref.[3] and ref.[4] have provided the main aerodynamic performance, stresses and control loads characteristics of this GAZELLE fenestron.
Ref.[5] surveyed the main features of the SA 360, 365 C and 365N DAUPHIN fenestron equipped with a 900 mm diameter fan (first flight in 1972): the technology is similar and the aerodynamic design is derived from the latest optimized version of the GAZELLE fenestron on which an extensive research test program had been achieved.
In 1980, studies were engaged to develop an advanced technology fan-in-fin concept with 1100 mm diameter rotor to be flight tested on DAUPHIN. So, the shroud, the fin and the blades have been fully redesigned with use of composite materials, ref.[6]. The new moulded plastic blades are cantilevered at two stations on plastic self-lubricating pitch bearings and linked to the hub with a unidirectional Kevlar fiber

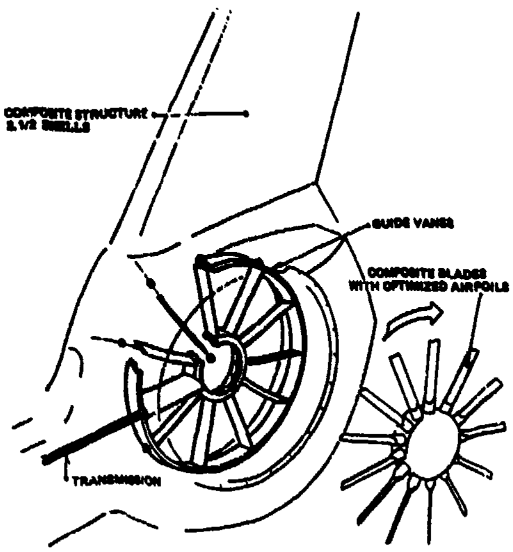

Fig.-2- Light helicopter composite fan in fin

spar providing low torsional rigidity for blade pitch variations.
This new "composite fenestron" is now fitted to the 365N1 DAUPHIN and 366G1 DAUPHIN, COST-GUARD version as well as on the 365M PANTHER prototype.

The most advanced technology is under study for light helicopters and includes new composite blades with optimised airfoils, stator blades in the diffuser replacing the gearbox support arms so as to recuperate the flow rotational energy. The shroud and the fin will consist of two half-shells made of composite structure, FIG.2.

3.0 OPERATIONAL ADVANTAGES

The various advantages of the fenestron have been presented in details in the above mentioned papers and reviewed in ref.[1]. We will simply recall the major points.

3.1 MANOEUVRABILITY, EFFICIENCY

In addition to all the flight tests and the whirl rig tests which have been achieved on the fenestron, more than 1700 hours of testing has been performed in the wind tunnel on 1/2 to 1/8 scaled models in order to get a in-depth understanding of its aerodynamic characteristics.

• HOVER

Due to the complexity of the flow environment of the tail rotor, much disappointment has been encountered in the past by helicopter manufacturers in sizing conventional tail rotors and consequently, by the pilots in using aircraft affected by poor yaw performance and handling. This explains why great efforts have been made to attempt a good understanding of this interactional aerodynamics-related topic, ref.[7] and ref.[8].

In hover with sidewind it is generally considered that in the wind direction-wind intensity map (FIG.3), three zones can be critical on conventional tail rotors:

- zone 1, RH sidewind (main rotor turning counter-clockwise): it is the maximum thrust critical zone which, under the most severe conditions of altitude/temperature including the yaw manoeuvering capability, determines the maximum disc and blade loading required for the tail rotor. In this case, the fin or the transmission fairing interacts the tail rotor creating flow blockage and whatever the selected solution, tractor or pusher tail rotor, there is a loss in the tail rotor net thrust. The fenestron is free of this interference. Furthermore, without intermediate gearbox, its smaller size and its lower position relative to the main rotor makes it free of adverse main rotor interaction.

- zone 2, aft sidewind: in ground effect, there is a combination of aircraft height and aft wind which tends to locate the ground vortex on the tail rotor. Due to the direction of rotation of this ground vortex, and exactly as for a conventional tail rotor, the blade bottom aft direction of rotation is unfavourable and the bottom forward direction of rotation has to be selected.

- zone 3, LH sidewind (main rotor turning counter-clockwise): in LH sidewind, the tail rotor flow opposes the wind and can enter the vortex ring state or recirculation mechanism resulting in pedal reversal or in erratic thrust response and large pedal activity. The T/R disc loading of the tail rotor is the critical parameter. Ref.[9] concludes that with a bottom forward direction of rotation the vortex ring state is retarded, and "that

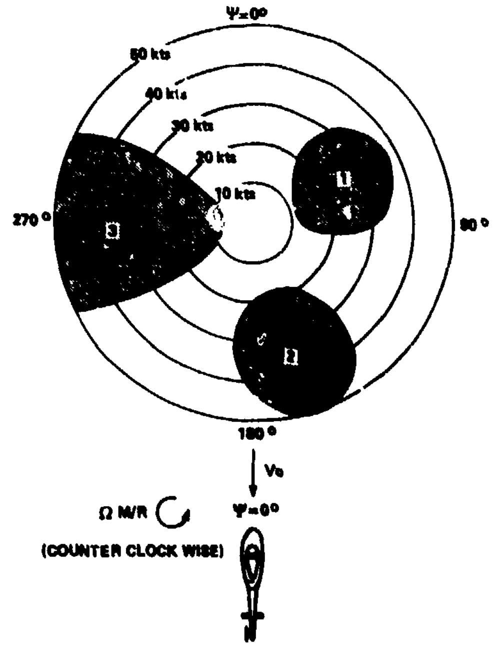

Fig.-3 Hovering critical zones for conventional tail rotors

larger helicopters with higher main rotor disc loading optimize with a tail rotor loading that permits good left sideward flight qualities up to 35 kts. For smaller helicopters, or those where minimum power to the tail rotor was the major consideration, left sideward flight up to 35 kts is not possible without large right pedal excursions". Assuming that the fenestron rotor diameter is half the equivalent tail rotor diameter, the momentum theory indicates that the mean induced velocity will be $2\sqrt{2}$ or 2.8 higher on the fenestron for the same anti-torque thrust. So, especially for light helicopters, the fenestron is very advantageous in left sidewind, opposing the wind direction. The bottom forward direction of rotation is also favourable to delay the flow recirculation phenomenon occurrence, as in the case of zone 2, aft sidewind.

Helicopters equipped with the fenestron have proven smooth handling and excellent yaw manoeuvrability: for example, the Coast Guard version of the Dauphin has demonstrated to be able to reach a 22°/sec yaw rate after 1.5 sec, in 35 kts left sidewind (main rotor turning clockwise) under critical altitude/temperature conditions at maximum gross weight.

FORWARD FLIGHT

In cruise flight, in order to get the best lift-to-drag ratio of the tail vertical surfaces, it is preferable to fully unload the fenestron. So, all the anti-torque thrust required has to be supplied by the fin which is of relatively large area. It is set at a given angle of attack with respect to the aircraft centerline and has a cambered section Consequently, the required power by the fenestron is extremely low as it only consists of the profile power which corresponds to one half the conventional tail rotor profile power in proportion with the blade area ratio.

The unloading of the fan in cruise has several other positive consequences as for instance:

- minimizing strains on all the rotating parts of the fenestron,
- or the capability of flying and landing with the tail rotor inoperative in case of failure.

Tests have shown that the directional stability depends almost entirely on the tail vertical surfaces sizing and very little on the fan. In early designs, it has been felt that the yaw control efficiency was poor in cruise within a three degrees sideslip zone, corresponding to a "deadband" appearing for neutral position of the pedal. Wind tunnel tests have provided a comprehensive analysis of this problem which is not relevant to the fan-in-fin concept in itself: the overall directional stability depends on the isolated fenestron stability (without fin) combined with the airframe directional stability. Tests clearly indicate that, as isolated, the fenestron is stable in yaw with no peculiarities, FIG.4. The problem is related to airframe stability and to wake effect due to the main rotor head and fuselage, reducing the control efficiency of rear surfaces. So, with the fenestron, it is necessary to improve the fuselage yaw stability if possible by reducing the wake effect, by improving tail surface efficiency and possibly by adding endplates on the horizontal stabilizer which can easily be adjusted during the development process of the aircraft, ref.[10].

3.2 SAFETY AND VULNERABILITY

In addition, the shroud naturally protects the rotor against external agressions and originally, the concept has been developed for the safety purpose. In fact, it remedies almost all drawbacks specific to conventional tail rotors.

It is the reason why for about two million flight hours have already been logged on helicopters fitted with fan-in-fin rotor, there has not been a single serious accident due to the fan-in-fin concept. This has to be compared with the above mentioned rate of helicopters crashed due to failed or impacted conventional tail rotors, which is in the order of 0.15 per 10,000 hours of flight as reported in the accident log book.

As illustrated on FIG.5, enclosed and sheltered in the duct, the fan cannot hit ground obstacles whatever the helicopter evolutions are. In flight, it is difficult, if not impossible, to have the fan hit by elements detached from the helicopter structure or from main rotor blades such as snow packs, ice accretions,..., or to catch cargo slings or hoist cables. Furthermore, when the aircraft is grounded, and the tail rotor operating, people can see the shroud and are not able to be injured by the shroud.

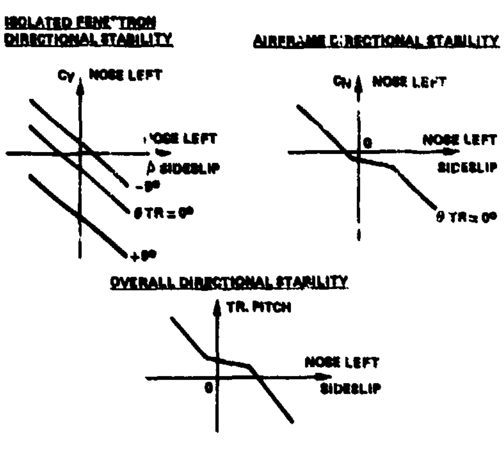

Fig.-4- Yaw stability in forward flight

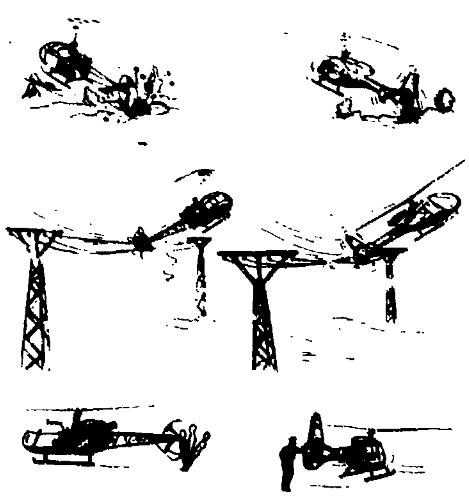

Fig.-5- Operational safety improvement by fenestron

The tail rotor is almost kept away from almost all possible external agressions.

As reported in ref.[1] and as demonstrated by numerous tests, this gives several advantages over the conventional tail rotor as far as sand or rain erosion is concerned in forward flight. Under snow or icing conditions, tests have also shown a better behaviour. In hover, and at low forward speeds, provisions must also be made, as for a conventional tail rotor for sand or dust protection, but due to higher centrifugal forces, ice accretion does not show up on the blades.

The experience shows that mean time between removal on tail rotor blades on the whole fleet of Aerospatiale Helicopters is about three times higher for fan-in-fin concept than for conventional tail rotor. It seems to need no special equipment for icing conditions as it has been experienced during numerous flight hours performed in these conditions.

Vulnerability tests have been undertaken which show that no serious damage occurs when 7.5 mm cartridge casings are thrown into the fan, and pellet impact of 7.5 mm caliber on a blade has practically no effect on the fan operation. It has been further shown that due to the large number of blades, the loss of one blade does not result in an immediate loss of the rotor, as it is generally the case for conventional tail rotors.

3.3 NOISE AND DETECTABILITY

It has been demonstrated, ref.[1], that the fenestron radiates less noise than the conventional tail rotor. Furthermore, the noise attenuation with distance is normally stronger than for conventional tail rotor, as the noise fundamental frequencies are higher by an order of magnitude approximately. Visual detectability when the helicopter is on watch, hiding behind tree lines is reduced in most cases (the conventional tail rotor will emerge from tree tops line but not the fenestron). Finally, reduced radar detectability can be obtained by the use of appropriate composite materials for the structure and for the short dimension blades which could use organic materials for anti-erosion protection devices.

4.0 AERODYNAMICS OF THE FENESTRON IN HOVER

Fan-in fin design criteria are set to provide for a given diameter, maximum thrust capability in hover with a high figure of merit.

From a pure performance point of view, the shrouded rotor is very attractive as, from momentum theory, (see momentum theory in ANNEX, as applied to the shrouded rotor), it offers for the same rotor disc diameter a power saving of about 30%, while developing the same thrust. The total figure of merit of the shrouded rotor can be expressed as follows:

$$F_m = \frac{T^{3/2}}{\sqrt{2\sigma}\,.\sqrt{2\rho S}\,.W}$$

σ being the ratio of the wake to rotor disk area, and S the area of the rotor disk.

How can the shroud improve the rotor efficiency?

The shroud improves the rotor efficiency because it can support the entering flow dynamic pressure, which gives it a thrust component as great as the rotor thrust.

This unloads the rotor which has less head pressure to generate for a given total thrust, and increases the mean depressure above the rotor disc.

The wash immediately downstream of the rotor is no more overpressured, as it would be on a free rotor and consequently does not contract.

In these conditions, the wake expansion is estimated as 1 D (D being the rotor diameter) as compared to 0.7 D, from Froude theory, for the conventional tail rotor. So, the thrust is shared as one half for the shroud and one half for the fan.

Furthermore, the diffuser even permits a slight depressure to be settled immediately downstream of the rotor and a slight expansion of the wake.

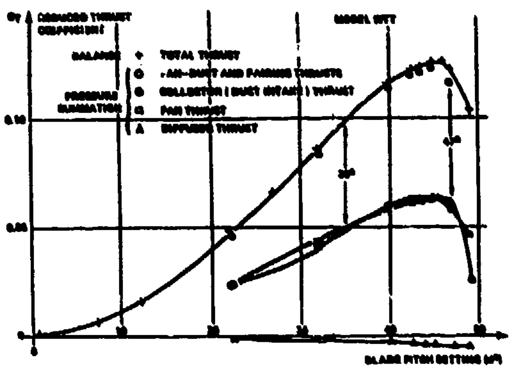

Fig.-6- Thrust sharing between fan and duct fairing

The measurements are well in agreement with this general theory.

FIG.6 shows the thrust versus pitch setting of a fenestron with a typical twist of -7°, as measured on a half-scaled model of the Gazelle fenestron. The fan thrust is derived from total pressure integration downstream of the rotor blades. The collector and diffuser thrust are derived from static pressure integration on the shroud, which is presented on FIG.7 for two values of the 0.7R pitch setting: 35°, which is characteristic of the current regime and 47° which is just beyond stall which occurs at 45° on this model fenestron. As previously explained, let us note that the flow is always depressed within the shroud. On the collector lips, high depressure levels are reached corresponding to maximal local flow velocities. The pressure profile depends on the curvature of the lips which determine the streamline curvature and the local depressure level. Immediately after the depressure peak, the flow has to face an adverse gradient which can result in a separated zone in the front of the rotor tip if the lip is not well rounded or if the collector length is too short.

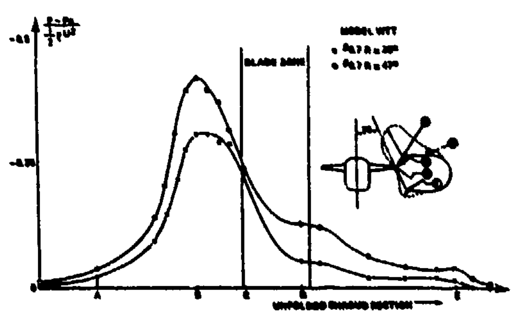

Fig.-7- Hover static pressure survey along the shroud

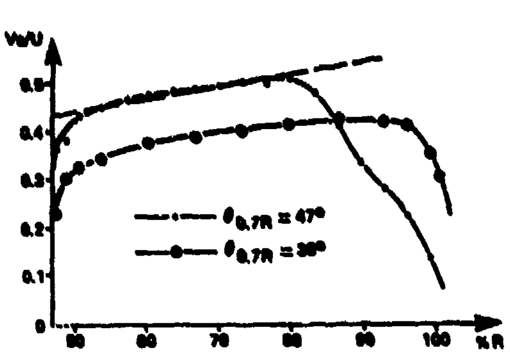

Fig.-8- Axial velocity distribution downstream the rotor blades

Velocity profile measurements have been achieved immediately downstream of the disc with a five-hole pressure probe which can give - after adequate calibration - total pressure, static pressure and airstream velocity components.

On FIG.8 and again for the same two characteristic values of pitch setting, the velocity profiles have been plotted. They illustrate the stalling mechanism of the fenestron: when the airfoils at the tip reach their Clmax, they cannot supply head pressure any longer to activate the flow in this area. The velocity profile is thus altered near the shroud and can no longer depress the inlet lip which limits the collector thrust.

On FIG.9, the flow rotational angle are also plotted for 35° and 47° pitch setting angles. So, the flow rotational angle β gradient generally varies as the axial velocity profile. At θ = 35°, the mean value of the rotational angle is about 10° whereas at θ = 47°, it is increased up to 18° in the potential flow zone. Close to the shroud, a large variation in the β angle up to 50° is noted. This corresponds to a viscous separated flow zone where the axial velocity vanishes due to blade section losses at stall. The flow rotation is due to the cascade deviation angle of the blades which increases as rotor solidity and blade camber. If it is not straightened, it corresponds to an energy loss. Considering these measurements, the idea came out to implement stator blades in order to convert the flow rotational energy into a pressure creating the additional axial thrust.

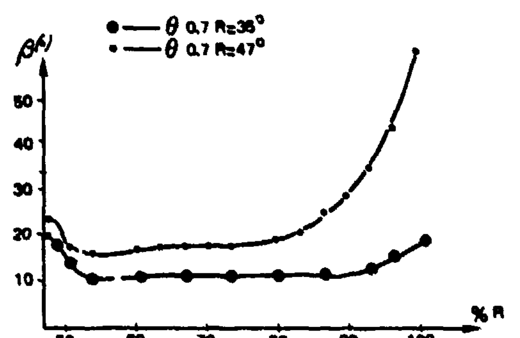

Fig.-9- Flow rotational angle downstream the rotor blades

5.0 FENESTRON CALCULATION METHODS

Two calculation methods are generally used by Aerospatiale.

The first one is directly derived from the local momentum and blade element theory, where the rotor disc is modelled with elementary independent rings. The airfoil characteristics and the local pitch angle are tabulated. It computes the axial and tangential velocities from axial thrust momentum equation and torque momentum equation. The shroud is globally considered in setting a given flow contraction σD from the rotor disc to infinity. This σD is derived from tests and is close to 1, in agreement with general momentum theory presented in Annex. This method is generally in use for performance estimation and sizing purpose.

A more advanced theory has been developed by METRAFLU (ref.[11]) on the basis of a compressor calculation code. This method accounts for the shroud shape interaction on the rotor. It is a quasi-tridimensional method in so far as the actual 3D flow is replaced by two bidirectional superimposed flows (FIG. 10):

- In the circumferencial plane (cascade airfoil calculation); this calculation is made with reference to tables, the NACA correlations issued from a great number of experimental tests on cascades.
- In the meridian plane; in this case, the calculation method uses a matrix resolution method, with an equation discreteness through finite differences. The flow is assumed not to be viscous, to be rotational, compressible and axisymmetric. The basic equations are the classical fluid mechanics equations (momentum, continuity, energy and perfect gas state equations).

Modifying these equations with additional relation results in:

$$\frac{\partial^2 \Psi}{\partial x^2} + \frac{\partial^2 \Psi}{\partial y^2} = Q(x,y)$$

Solving the above equation for every axial station allows calculating the flow within a meridian plane and requires data issued from circumferential plane calculation for a given radius. The tridimensional flow is restored by combining both bidimensional calculations in an iterative way.

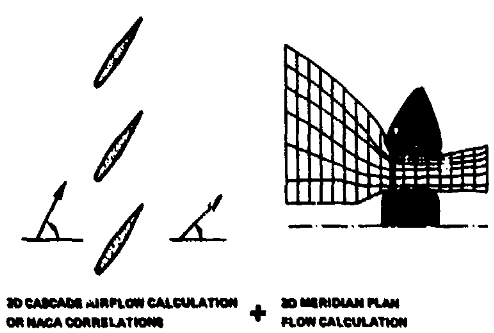

Fig.-10- Fenestron calculation method (METRAFLU)

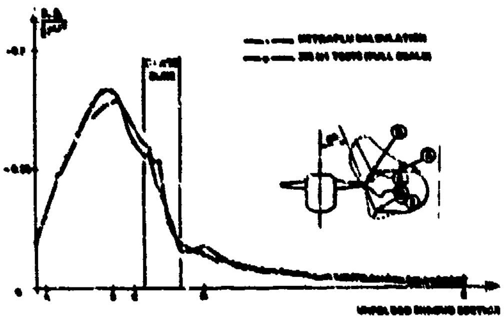

Fig.-11- Static pressure computation on the shroud

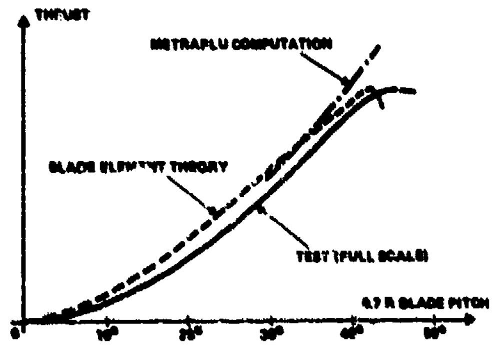

Fig.-13- Predicted and measured fenestron thrust versus pitch characteristic

These theories have been correlated with test results which have been obtained at the bench at scale 1.

FIG.11 shows correlation obtained with the METRAFLU method on static pressure measurements on the shroud in a meridian plane, at moderate pitch angle setting. Upstream of the depressure peak, the flow modelling is not exactly in accordance with the shroud shape, due to the calculation method assumptions and, the computed results have not been reported. This does not influence the downstream results where the prediction is quite correct, even in the diffuser. In particular, the maximum depressure peak and the pressure recovery gradient are correctly predicted. Some local discrepancies are noted at the blade zone. They are due to blade tip vortices which are not taken into account in the potential calculation.

FIG.12 show the predicted and measured axial and tangential velocities with the two methods. The viscous effects due to blade tip vortices result in a boundary layer development close to the hub and the shroud which are not computed. It results in a flow blockage which increases the axial velocity in the non-viscous zone. This explains why the axial velocities computed values are underestimated. In the case of the local momentum blade element theory, the axial velocities are a little more underestimated. This is partly due to the fact that computed 2D airfoil characteristics have been used instead of tests values which were not available at the moment. The tangential velocity correlation is generally good for both methods.

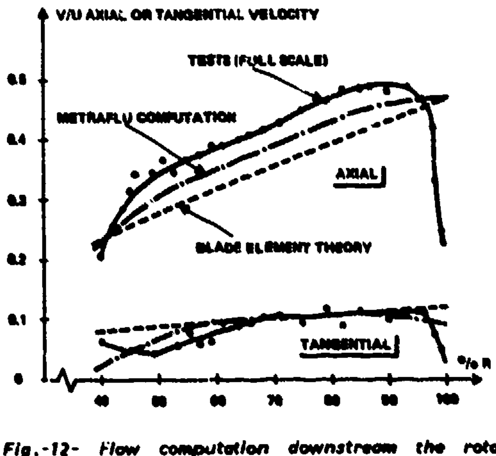

Fig.-12- Flow computation downstream the rotor blades

The predicted and measured thrust versus pitch angle characteristics are compared on FIG.13. Note that the local momentum blade element theory seems to correctly predict the thrust stalling level, although 2D airfoil characteristics are computed values with estimated stall. The METRAFLU method is a potential method and cannot give accurate information after stalling. The stalling can be estimated from the calculated spanwise load factors on the blade.

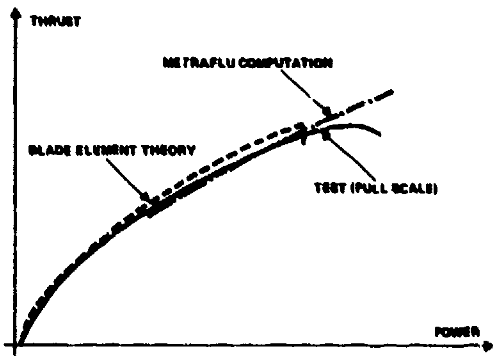

Fig.-14- Predicted and measured fenestron thrust versus power characteristic

The predicted and measured thrust versus power characteristics are compared on FIG.14. The METRAFLU computation gives quite good results before stall. The local momentum theory gives acceptable results, considering the use of 2D airfoil computed characteristics.

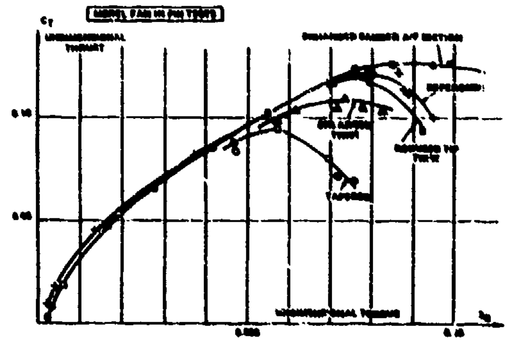

Fig.-15- Thrust / torque relationship for different blade fan geometries

6.0 PERFORMANCE

To improve hover performance, the following parameters have been studied:

- Blade planform, twist, and airfoil sections camber as shown in FIG.15 from earlier tests on wind tunnel model.

- Effect of new airfoil sections, specifically developed in co-operation with ONERA and Aerospatiale, so as to increase maximum lift capability at different blade spanwise sections as represented in FIG.16. This new fenestron airfoil family with spanwise variable relative thickness has essentially been designed with a view to increasing the load at the blade tip, so as to get the maximum depressure level on the shroud and delay on as far as possible the blade tip stall.
  An other new rotor with further increased camber blade sections has been tested which gave even more thrust at stall. But as the required power at zero thrust, which is important in forward flight, was much higher, due to higher section drag at zero lift, the test results have not been reported.

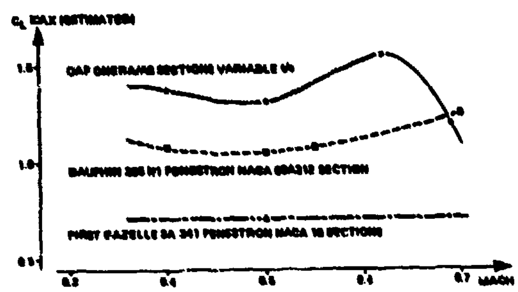

Fig.-16- Fenestron airfoil clmax improvement

- Effect of diffuser angle and static vanes, set downstream of the fan, to improve the flow expansion (higher σD) and to straighten the airflow in order to recover the flow rotational energy as presented in FIG.17. The diffuser angle α is actually limited to a practical angle value of about 10°, as with higher diffusion angles, flow instabilities may occur as interacted with the main rotor. This effect had been evidenced on early versions with the bottom aft fenestron direction of rotation which had been forsaken because of poor performance in rear wind in ground effect.

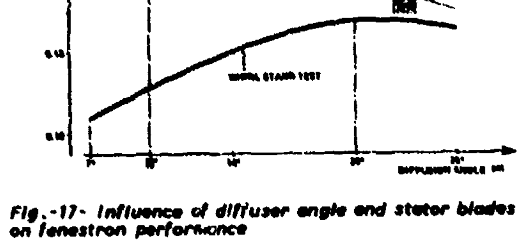

Fig.-17- Influence of diffuser angle and stator blades on fenestron performance

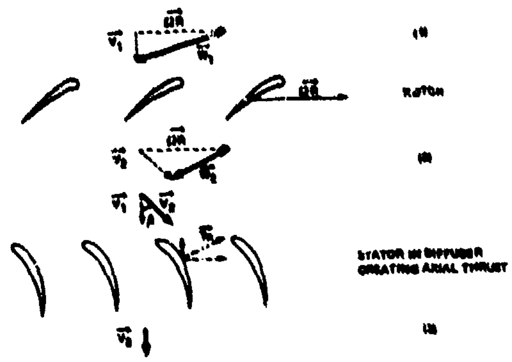

Fig.-18- Influence of stator blades on axial thrust

- The effect of stator blades is illustrated in FIG.18. In the rotating plane, the flow is deviated by the rotor blade. The deviation angle increases with blade solidity and camber. It results, just downstream of the rotor blades, in an absolute velocity V_2 angle β relative to the axial direction. The stator blades deviate the flow from V_2 to V_3, directly creating an axial thrust and some pressure recovery as the velocity slightly decreases.
  FIG. 19 shows (full scale measurements) that the flow has been almost completely straightened with these stator blades.

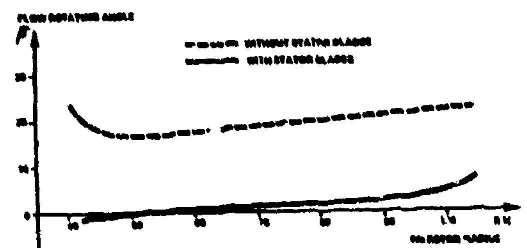

Fig.-19- Influence of stator blades on flow rotating angle at the diffuser exit

The improvements obtained on separate modifications briefly reviewed above, have been integrated on a scale one fenestron research test bench presented in FIG.20, for the Dauphin N1 helicopter. This test

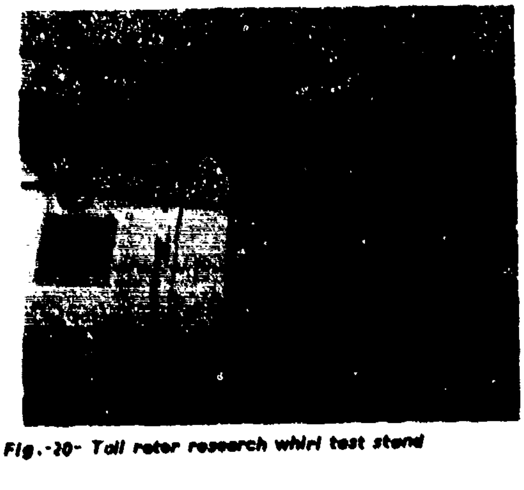

Fig.-20- Tail rotor research whirl test stand

facility allows for accurate measurements of tail rotor performance, as well as pressure survey and noise radiation measurements. Several types of blades, duct geometries, and guide vane setting angles have been recently evaluated.

FIG.21 presents figure of merit data as a function of the mean blade loading coefficient Czm, obtained by direct on line data processing at the test bench site which allows to obtain precise data in the complete thrust domain of the fan-in-fin. Each characteristic is presented with least square curve fitting on about 150 test values.

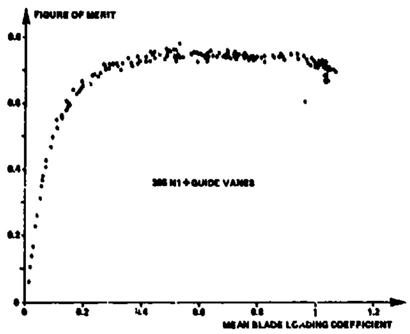

Fig.-21- On line data processing of whirl test stand data

As presented in FIG.22, maximum figure of merit can be increased by 7% and maximum thrust by 37% as compared to the present production 365 N1 Dauphin fan-in-fin due to guide vanes (or stator blades) and new airfoil section shapes for the fan blade. Furthermore, the figure of merit stays quite constant for large mean lift coefficient (or thrust) of the fan-in-fin. Substantial efficiency improvements are shown in FIG.23 compared to current conventional tail rotor with two- or four-bladed design using new airfoil sections technology.

	CONFIGURATION	$(FM)_{max}$	$(C_{Zm})_{max}$
1	365 N1 (REFERENCE)	0.71	0.825
2	WITH GUIDE VANES	+4.2%	+26%
3	WITH OAF AIRFOIL SECTIONS + G. VANES	+7.0%	+37%

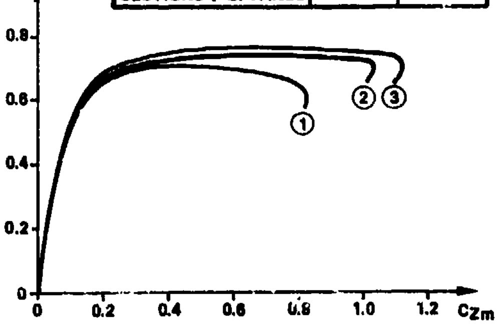

Fig.-22- Fenestron performance improvements (full scale ground tests)

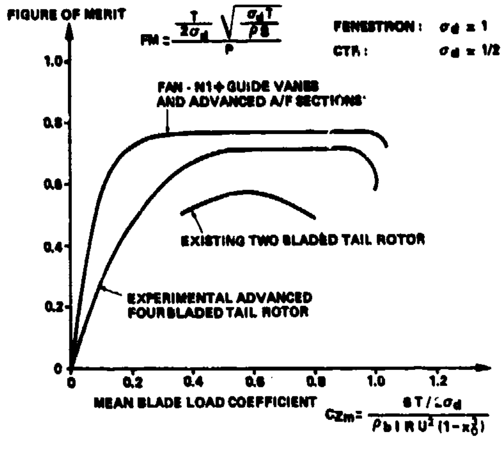

Fig.-23- Isolated tail rotor efficiency

It is to be noted that comparisons between conventional and fan-in-fin tail rotor performance should take into account not only the isolated tail rotor efficiencies - as shown in FIG.23 - but also the fin blockage effect normally present on conventional tail rotor. This effect is illustrated in FIG.24, which presents for a given tail rotor power the equivalent fenestron/classical rotor diameter ratio as a function of figure of merit ratio and fin blockage in percent of thrust. In particular, for an improved figure of merit of 30% (FM ratio of 130%) and 5% fin blockage effect, an equivalent fan-in-fin would have half the diameter of a conventional tail rotor.

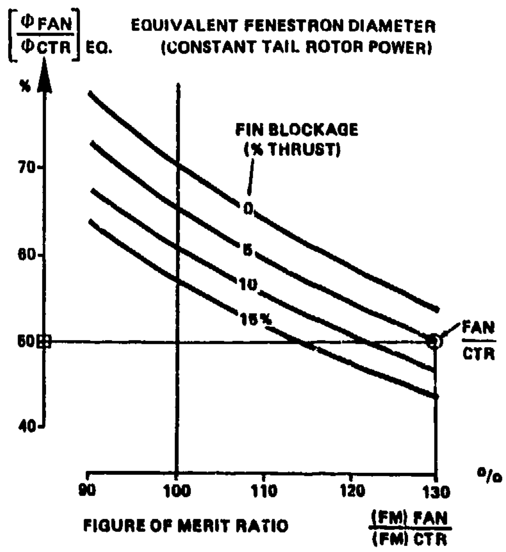

Fig.-24- Determination of fenestron / conventional tail rotor equivalent diameter at constant power

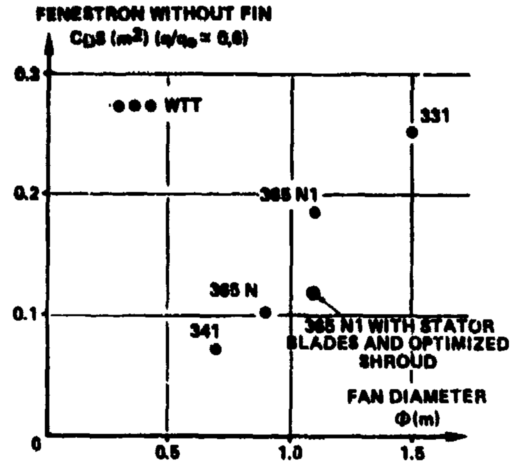

Fig.-25- Fenestron drag

The implementation of the stator blades in the diffuser has also improved the pressure recovery and tests have been completed with a reduced diffuser length, resulting in a narrower shroud. The tests have demonstrated that up to a certain limit, it does not affect the performance. This finally results in lower drag of the fenestron. The drag saving is estimated to be as high as 40% on the Dauphin 365N1 fenestron. FIG.25 compares these drag values with the drag of various Aerospatiale helicopter fenestrons, without fin, and assuming that the dynamic pressure is reduced to 60% of the freestream dynamic pressure due to fuselage and main rotor hub wake.

7.0 CONCLUSION

The fan-in-fin or fenestron concept has been originally developed for the only sake of improved safety and at an accepted penalty of weight, required hover power and cost.

The operational experience shows that the improved safety was indeed demonstrated, as no major accident occurred due to fenestron problems on nearly thirteen hundred fenestron-equipped helicopters, which have been flown for more than two million hours.

In addition to the research and development work conducted for eighteen years, recent research work at scale-one bench in hover have enabled new performance gains with optimized airfoils and stator blades in the diffuser, as well as the possibility of reducing the shroud width, without hover performance penalty, which results in drag saving in forward flight. Two calculation methods have been correlated on this tests giving quite good performance and flow predictions in hover.

This has brought the fan-in-fin concept to a level which makes it attractive, as compared to the classical tail rotor, on nearly all points of comparison for light- and medium-weight helicopters:

As regards performance: fan-in-fin with equivalent effectiveness can be designed with a diameter almost half the classical tail rotor diameter, due to aerodynamic improvements on airfoil shapes, duct geometries and stator blades.

As regards handling qualities: appropriate choice of fin geometry and size, and duct geometry provides better handling qualities.

As regards overall weight and cost: the fan-in-fin concept developed with advanced composite technology is equivalent to the latest conventional tail rotor for light helicopters and shows substantial reduction in weight and cost when compared to tail rotor mounted on top of a tail pylon.

Considering the complementary advantages of improved safety and reliability, reduced detectability and vulnerability, the "fenestron" fan-in-fin concept can presently be considered as the best anti-torque system for the single main rotor light and medium size helicopters.

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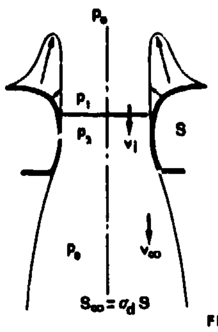
ANNEX 1

MOMENTUM THEORY



BERNOULLI EQ : $\begin{cases} P_1 + \frac{1}{2}\rho v_i^2 = P_0 \\ P_2 + \frac{1}{2}\rho v_i^2 = P_0 + \frac{1}{2}\rho v_\infty^2 \end{cases}$ (1)

MASS CONSERVATION : $\rho S v_i = \rho(\sigma S) v_\infty$ (2)

MOMENTUM EQ : $T = T_{ROTOR} + T_{SHROUD} = (\rho S v_i)\, v_\infty$ (3)

ENERGY EQ : $P_i = (\rho S v_i)\frac{1}{2} v_\infty^2$ (4)

FROM : (1) $T_{ROTOR} = (P_2 - P_1)\, S = \frac{1}{2}\rho v_\infty^2 S$ (5)

(3,5) $\frac{T_{ROTOR}}{T} = \frac{v_\infty}{2 v_i} = \frac{1}{2\sigma}$ (6)

(2,3) $v_i = \sqrt{\frac{\sigma T}{\rho S}}$ (7)

(4,5,6) $P_i = T_{ROTOR}\, v_i = \frac{T}{2\sigma}\sqrt{\frac{\sigma T}{\rho S}} = T\sqrt{\frac{T}{4\sigma\rho S}}$ (8)

ROTORCRAFT DESIGNS FOR THE YEAR 2000

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SUMMARY

Many different types of rotorcraft and higher disk loading Vertical Takeoff and Landing (VTOL) concepts have been investigated in the past, but none have survived to production except for helicopters and one direct-lift turbofan VTOL. With modern technology developed in the past two or three decades, some of the earlier concepts might now be more practical and new concepts have become feasible. This paper examines some of the rotorcraft concepts that can offer higher speeds than the pure helicopter, including the compound helicopter, ABC, tilt-rotor, X-Wing, and stowed rotor configurations. All of these are potentially practical aircraft. Economic considerations will dictate that the pure helicopter is here to stay; higher speed rotorcraft are not as cost-effective at short ranges.

Also examined is the approach to helicopter sizing for heavy-lift applications. It is concluded that a super-large helicopter, sized to carry the heaviest payload needed, is unaffordable. A better, more cost effective solution is to develop a moderately large helicopter that will carry most of the required loads, and to use the twin-lift technique to transport the occasional extra-heavy load.

INTRODUCTION

The spectrum of heavier-than-air vertical takeoff and landing aircraft may be divided into two broad categories on the basis of disk loading. Rotorcraft, using exposed rotating blades and with disk loadings typically on the order of 5 to 15 lbs. per square foot (approximately 25 to 75 kg/m^2), represent the end of the spectrum that is currently the domain only of the helicopter. Higher disk loading VTOL's, such as tilt wing/propeller and direct-lift turbofan configurations, represent a different class of aircraft in that the hover downwash becomes greater than humans or most vegetation can tolerate; the ability to hover for extended periods is severely curtailed; the near-field noise typically becomes excessive; achieving adequate control power becomes more difficult; and the possibility of helicopter-type safe autorotative landings in case of total power loss disappears. Thus, the higher disk loading class of aircraft will be limited to missions and operational requirements that are different from those typical of current helicopters. A large number of different configuration types are possible within these two classes, and many of each class have been explored in the past, with wind tunnel model tests, flying test beds, and even a few prototype production aircraft. This effort was conducted largely during the late 1950's and 1960's, when there seemed to be considerable emphasis on establishing alternatives to the helicopter that would have much higher speed capability. With one notable exception (the Hawker P1127, which evolved to the present-day Harrier), the large investment in these endeavors, totalling in the hundreds of millions of dollars, failed to produce any aircraft that actually reached the production stage. There were many reasons for this lack of success; the two most common were a missing element of technical feasibility, most often inadequate control power or stability in hover and low speed flight; and the lack of enough payload to make the concept economically attractive.

In the 20 to 30 years that have passed since most of the experimentation took place with "advanced" configuration concepts, there has been significant technological progress on a number of fronts that bear on the feasibility or economic viability of these concepts. The empty weight fraction of all configurations can be reduced substantially by the use of advanced composite materials now available. Smaller but significant weight savings are also available with more modern engines, transmissions, and control systems. Weight is critical to the relative merits of different configurations, and changing technology can alter the relationship. Consider two configurations of older technology: a helicopter with an empty-to-gross weight fraction of 0.70 and a high speed rotorcraft (with bigger engines, wings, propellers, etc.) with an empty weight fraction of 0.90. Allowing five percent in each case for fixed useful load (pilots, trapped fuel, etc.), leaves .25 and .05 respectively for maximum payload fraction. This five-to-one advantage of the helicopter (at zero range; greater at finite range) overwhelms the speed advantage of the "advanced" type; there is really no contest. Now consider the same two configurations with the empty weight fraction decreased by 20 percent by the use of modern composite structures, etc. With the same fixed useful load allowance, the zero range payload fractions increase to 0.39 for the helicopter (a 56 percent increase) and to 0.23 for the high speed vehicle (an increase of 360 percent). Although the helicopter retains a payload advantage, the higher speed of the other vehicle, depending on circumstances, could more than offset this. For example, if the speed is twice as high, the productivity in a transport mission (payload times speed) becomes greater than for the helicopter.

Other technology factors also affect feasibility or economic viability. There have been substantial improvements in airfoils and blade tip designs for rotors, which improve hover figure of merit, maximum lift, and maximum allowable Mach numbers. The microelectronic revolution also has an impact; modern digital flight control and stabilization systems are now available that could solve some of the control problems encountered in the past.

Thus, configurations once tried and found inadequate may now be revisited. Some of them are likely to be attractive candidates for future development. This paper will review some of the high speed rotorcraft concepts considered to be good candidates for future development. It will also consider the problem of the most effective solution to heavy lift helicopter requirements of the future. The possible future of

the high disk loading types of VTOL aircraft is not considered here, except to say that the fundamental disadvantages of high disk loading, previously mentioned, will continue to detract from their capabilities and applications.

HELICOPTER SPEED LIMITATIONS

For many years, helicopter speed capabilities increased steadily with time, as shown in the speed record history in Figure 1. These records represent the absolute maximum speeds, all of which are in the two shortest distance categories, in the unrestricted weight class of "pure" helicopters (i.e. no propulsion except from the rotor which also produces lift). The trend has leveled off in recent years, with no new record between 1978 and 1986. When this paper was written, a new record of approximately 216 knots (400 km./hr.) was claimed, but not yet certified by the Federation Aeronautique Internationale, for the Westland Lynx with an experimental rotor. This new mark is an impressive achievement, and is considered to be close to the maximum to be expected for helicopters. Normal operational speeds are lower than record speeds, of course. It is not anticipated that the pure helicopter will ever have a routine capability much above 200 knots (371 km./hr.).

The most recent speed records set by a Sikorsky product was established in 1982 by the S-76 helicopter, Figure 2. This aerodynamically clean, nominal 10,000 lbs. (4535 kg.) class helicopter set eight official world speed records in two weight classes, plus one in the unrestricted weight class: a speed of 345.7 km./hr. = 186.5 kts. over a distance of 500 km.

The reason pure helicopters are speed-limited is illustrated in Figure 3. Although the rotor is unexcelled for producing lift and propulsive force at low speeds, the capability drops substantially as forward speed is increased. The cause of this characteristic is the reduction in velocities relative to the blade on the "retreating" half of the rotor disk, resulting in large reductions in local dynamic pressures available for producing lift. The dynamics pressure on the "advancing" half of the disk are increased but are not usable with conventional rotors because of blade dynamic response and the need for roll trim. At 200 knots (371 km./hr.) the lift capability is typically only on the order of one-half that at 100 knots (185 km./hr.), and the propulsive force capability is typically reduced by a factor of five or more, whereas the lift requirement is essentially constant and the propulsive force requirement has increased by a factor of four. At some speed above 200 knots the propulsive force capability vanishes altogether.

THE COMPOUND HELICOPTER

A logical means of extending the performance envelope of the pure helicopter is by compounding, i.e. supplementing rotor lift by means of a wing and providing some means of auxiliary propulsion. A properly sized wing supplements rotor lift in a nearly ideal manner as shown in Figure 4. The wing lift potential increases with the square of the flight speed and the combined lift capability is quite flat up to 200 knots, beyond which it increases.

Many experimental compound helicopters have been built and flown, and two designs reached the production prototype stage. One of the earliest experimental compounds was the McDonnell XV-1, Figure 5. It had a wing, a pusher propeller and a pressure-jet driven rotor system with tip burning. Another experimental aircraft was the Sikorsky S-67 prototype gunship helicopter, Figure 6. It incorporated a wing but not auxiliary propulsion, and was the fastest helicopter not having auxiliary propulsion ever built by Sikorsky, setting a world speed record of over 191 knots (354 km./hr.) in 1970. It was also the most maneuverable. A research compound helicopter, the NH-3A, is shown in Figure 7. It was based on the Sikorsky S-61 but incorporated a wing, two turbojets for auxiliary propulsion, and airplane-type control surfaces. It was flown at speeds up to 230 knots and provided valuable data which confirmed the capabilities of the compound concept. The fastest experimental compound helicopter was a derivative of the Bell UH-1, Figure 8. A high ratio of installed jet thrust to weight allowed flight speeds up to approximately 275 knots (509 km./hr.)

One aircraft in the compound helicopter category that was planned for production in the past was the Fairey Rotodyne, Figure 9. This aircraft also used a pressure jet rotor with tip burning. Another production prototype was the Lockheed AH-56 Cheyenne, Figure 10, which used a shaft-driven rotor having blades without flap or lag hinges ("rigid" rotor) and a pusher propeller at the tail. Neither of these aircraft reached the production stage; reasons for stopping were many, but cruise speeds below initial expectations and increased costs were among the factors that influenced the final decisions.

A unique rotorcraft configuration that is sometimes classified as a compound is the Sikorsky Advancing Blade Concept or ABC®. Two rigid, counterrotating, coaxial rotors are utilized for lift rather than a single main rotor plus wing. The lift potential of the advancing blade may be realized because of the stiffness of the blades and the counterbalancing of the two rotors, Figure 11. Lift capability of the ABC increases with speed, unlike that of a conventional helicopter rotor. The concept has been proven by the XH-59A research aircraft shown in Figure 12. Again, two turbojet engines were employed for propulsion. This aircraft reached 240 knots in level flight and exceeded 260 knots in descent. The ABC provides a particularly compact and maneuverable vehicle that should be well suited to nap-of-the-earth operations or to an air-to-air combat role.

TILT-ROTOR AIRCRAFT

The tilt-rotor aircraft is a concept not usually associated with Sikorsky Aircraft, but we have made a number of serious studies of it. As shown in Figure 13, two lifting rotors, mounted from pods at the wing tips, tilt forward 90 degrees to convert the aircraft to a "conventional" airplane with oversized propellers, with the wing supplying 100 percent of the lift in cruise. The cruise speed and lift-drag ratio can exceed those of a pure helicopter by a significant margin. For long-range operations, it can have attractive characteristics. A more advanced variant of the tilt rotor is one which stops and folds the rotor blades in cruise, relying on other propulsive means, such as a convertible fan-shaft engine, in high speed cruise. A Sikorsky design of this type is shown in Figure 14.

An early experimental tilt-rotor aircraft was the Bell XV-3, Figure 15; a more modern counterpart is the Bell XV-15, Figure 16. This aircraft has achieved speeds of approximately 300 knots (556 km./hr.). Now in the prototype fabrication stage is the much larger Bell-Boeing V-22 Osprey, intended to be the first practical application of the tilt rotor concept.

X-WING

A relatively new configuration, just reaching the exploratory flight test stage, is the Navy X-Wing Concept. Under NASA and DARPA sponsorship, Sikorsky Aircraft is developing an X-Wing rotor, which will be tested extensively on the NASA Rotor Systems Research Aircraft, shown in Figure 17. An artist's concept of a possible military application is shown in Figure 18. The X-Wing utilizes a shaft-driven four-bladed rotor with extremely stiff blades. It takes off like a conventional helicopter but has auxiliary propulsion or convertible fan/shaft engines that will permit it to reach high forward speeds with the rotors turning. At a suitable conversion speed, on the order of 200 knots (371 km./hr.), the rotor is braked to a stop and positioned with two blades swept forward 45 degrees and two swept aft 45 degrees. The blades are symmetrical fore-and-aft and utilize pneumatic control of a thin jet of pressurized air out of the leading and trailing edges of the blade, as shown in Figure 19, to provide circulation control to maintain full rotor lift in all flight regimes. Photographs of one of the experimental blades, and of the pneumatic valving system in the hub for azimuthal control of the air supply, are shown in Figures 20 and 21 respectively. The X-Wing is potentially capable of flight speeds in the 400 to 500-knot range (741 to 927 km./hr.). This configuration would not be possible without some of the technology advances mentioned earlier -- specifically high modulus graphite composite materials to provide the required stiffness at acceptable weights, and a sophisticated quad-redundant digital flight control system to handle the pneumatic valves and other onboard systems that an unassisted human pilot would not be able to cope with. It is a good illustration of the point that advancing technology can change the relative merits of various concepts.

STOWED ROTOR AIRCRAFT

The ultimate in high-speed rotorcraft is probably the stowed rotor configuration, which uses a conventional wing to provide all lift in the cruise mode, after stopping the rotor and stowing it away in the top of the fuselage. Ideally, the wing is optimized for cruise and the overall drag is as low as for an equivalent size fixed-wing aircraft. Once the rotor is stowed, high subsonic speeds should be readily available; it should even be possible to design for supersonic capability if the mission demands it. Figure 22 shows a Sikorsky conceptual design of such an aircraft.

The stowed rotor concept has not yet reached the flight test stage. A large-scale model of a Lockheed three-bladed stowed rotor concept, Figure 23, was once tested in the NASA Ames 40-by-80-foot wind tunnel. Rotor stops and starts were achieved at wind tunnel speeds as high as 140 knots (259 km./hr.), but this test also demonstrated that blade aeroelastic deformations and aircraft pitch and roll moment disturbances were of considerable concern. Early Sikorsky scale model wind tunnel tests of a rotor designed for a stowed rotor configuration showed that the blade aeroelastic deformations that occur when stopping the rotor in flight could be extreme, constituting a critical problem for this configuration. These tests also revealed excessively large aerodynamic moments in simulated gust conditions that would upset the aircraft equilibrium and place severe demands on the elevator and aileron controls.

Increasing the design disk loading of a stowed rotor concept is one approach to reducing the aeroelastic and control disturbance problems, because this results in shorter but wider-chord blades. A much more positive solution, however, is to use an inflight variable diameter rotor which reduces blade length substantially prior to stopping, as shown in the conceptual model in Figure 24. The feasibility of this approach was established by Sikorsky with a four-bladed, nine-foot diameter dynamically-scaled model which could be varied rapidly in diameter at full rotational speed at the operator's discretion, down to about 5.5 feet. This model, shown in Figure 25, utilized a differential gear set inside the rotor head and a jackscrew inside each telescoping blade to achieve the desired control. At minimum diameter, the rotor was easy to stop or start without either aeroelastic or aircraft control disturbance problems. The model demonstrated diameter changes, stops, and starts at forward speeds up to 150 knots (278 km./hr.) true airspeed. Fore-and-aft blade positioning was also demonstrated to verify ease of folding for subsequent retraction into a fuselage. At minimum diameter with the rotor turning, tests were also conducted to 400 knots (741 km./hr.) true airspeed to investigate applicability to a very high speed compound helicopter configuration. The program, which was supported by the U.S. Army, was continued with a successful laboratory test of a full-scale blade jackscrew mechanism.

Although the variable-diameter rotor program results were positive, with essentially all technical objectives achieved, there was a reduced level of customer interest in any high speed aircraft at that time, and the program did not proceed to a flight test for that reason.

FUTURE TRENDS

As described in the Introduction, all configurations will benefit from modern technology, particularly from the use of advanced composite materials. It is likely that all of the concepts described would be viable for some missions compared to helicopters, but it is not likely that they are equal in overall merit. In the current economic climate, it is quite improbable that all of the configurations will be resurrected for a new attempt at developing an advanced vehicle for production. Although it is not the purpose of this paper to predict which configurations will be winners in the near future, a brief critique of each is offered.

Of the various high speed rotorcraft configurations, the one with the least technical risk is the compound helicopter. There have been about a dozen different compound configurations flown with at least four distinctly different rotor configurations (articulated, teetering, "rigid", and ABC). All were technically successful to various degrees. The risk of the compound is the one of economic viability. The speed potential is limited to about 250 knots (463 km./hr.), primarily because the drag of the

exposed rotor head makes it too inefficient at higher speeds. The drive train complications caused by the need for an RPM reduction at high flight speeds, to avoid excessive Mach numbers on the advancing blade, also contribute to it being less attractive beyond 250 knots. The weight of a wing and auxiliary thrust system reduces the payload; the added drive train components of the thrust system impacts reliability and maintainability. Weight is the chief concern; does the increased speed make up for the loss of payload? The answer in the past has always been: not quite. In the future, the answer will probably be yes. The compound has one large advantage over the other types, which is that nearly any existing helicopter can be compounded. It should be considerably more rapid and less expensive to develop a compound derivative of a production helicopter than to design an entirely new aircraft from the ground up, as required for the other types discussed. For this reason, it is considered likely that one or more compound helicopters will be developed to meet various needs in the future where maximum speeds of 200 to 250 knots (371 to 463 km./hr.) are adequate.

The tilt rotor concept appears to be alive and well, with the V-22 development program in progress. The tilt rotor unquestionably has some attractive attributes, and it is considered likely that this type will capture some portions of the rotorcraft market in the future. Some enthusiasts claim that the tilt rotor will displace helicopters almost completely. This is highly unlikely, because, like all other configurations, it has a few poor attributes as well as good ones. Like all of the high speed rotorcraft, it suffers a penalty in useful load fraction relative to the helicopter, and thus is handicapped at shorter ranges where its speed advantage may not be significant. Another drawback is its disk loading, which appears to be on a trend that is double that of conventional helicopters, as shown in Figure 26. As disk loadings increase, so do the downwash velocities. The side-by-side rotor arrangement also causes an interference effect when hovering in ground effect. Two pronounced fountains of air form and rise to considerable altitude, ahead of and behind the aircraft in the plane of symmetry. This flow will tend to throw sand and other debris into the rotors and to obscure the pilot's vision when operating from unprepared areas.

The tilt rotor has other limitations. Using the same rotors for lift in hover and for propulsion in cruise requires a compromise because of the large difference in thrust required for those two conditions. The rotor tends to be undersized for hover and much too large in cruise. The location of the rotors at the tips of a wing result in potential aeroelastic instability and vibration modes that must be avoided. The tilt rotor can accelerate through its conversion to cruise very rapidly, but is not nearly as good when decelerating back to a hover. It is cumbersome in low-speed maneuvers compared to most helicopters.

Despite the list of disadvantages, the tilt rotor is believed to be a good choice for missions where high speed and long range are frequent requirements. There are also modifications to the tilt rotor concept which can provide significant improvements. One of these is the use of a variable diameter rotor, as shown in Figure 27. This serves as a means of reducing disk loading down toward the more desirable conventional helicopter values, and eliminates the mismatch between hover and cruise thrust by reducing blade area and tip speeds without changing the RPM. While it adds complexity, the performance and other benefits are substantial.

The next configuration on the list is the X-Wing. This concept promises superior capabilities once it is fully developed. The X-Wing is introducing a number of advanced technologies, including ultra-stiff composite rotor blades, circulation control aerodynamics with both leading and trailing edge blowing, multicyclic pneumatic vibration control, a quad-redundant digital flight management system, the necessary systems for in-flight rotor stops and starts, and several other novel design features. The concept development program is now reaching the flight test stage; production prototype development is expected to follow. An artist's sketch of an advanced X-Wing, available some time after the year 2000, is shown in Figure 28.

The future of the stowed rotor is uncertain at this time, because there are no currently active development programs. From the long-range viewpoint, however, it seems probable that it will eventually be developed as an ultra high speed, low disk loading VTOL. The success of the variable diameter rotor concept is believed to be a key element; all other aspects are relatively straightforward. A possible design of this aircraft type is shown in Figure 29. Like the rest of the concepts, the stowed rotor has its share of drawbacks. In addition to the weight concerns shared with the rest, it requires a relatively high degree of mechanical complexity, with a corresponding concern for reliability and maintainability. Because the rotor is not used at all in cruise, it is also not accumulating any fatigue cycles for the largest portion of its flight time. Whether this fully compensates for the added parts is uncertain. The trend in fixed-wing aircraft is that mechanical complexity has increased steadily over the years; this seems to be the price of improved performance. If this is true for rotorcraft, then the stowed rotor could be very successful.

All of the high-speed rotorcraft configurations still have to compete with the conventional helicopter for most mission applications. This will not be easy to do, as discussed briefly in the Introduction. Although it is true that the air vehicle empty weight is reduced substantially by composite materials and other advanced technologies, there is also constant pressure from the users, particularly military customers, to add special equipment packages and features to enhance the mission capabilities, improve reliability and maintainability, decrease vulnerability and detectability, improve crashworthiness, etc. These demands all add weight, tending to negate the technology improvements that lowered the basic empty vehicle weight. The weight savings benefit the high speed vehicles the most; features that add weight will penalize them more than the helicopter.

A comparison of typical payload fractions for an advanced 300-knot (556 km./hr.) rotorcraft, a 180-knot (334 km./hr.) helicopter of comparable technology level, and a 500-knot (927 km./hr.) conventional airplane is shown in Figure 30. Relative to the airplane, both of the others suffer a payload penalty at zero range, with the faster vehicle having the larger penalty. Because the pure helicopter has a cruise efficiency (equivalent lift drag ratio) significantly less than the airplane, the additional fuel consumed results in the payload penalty, relative to the airplane, increasing with range as shown. The 300-knot rotorcraft is assumed to be more efficient, approaching the lower fuel consumption of the airplane, so the payload difference between the two rotorcraft narrows with increasing range.

For any transport mission (delivering people or cargo), an important measure of effectiveness is productivity, defined as payload times block speed. Because large aircraft can carry more payload than small ones, it is necessary to divide productivity by aircraft weight to determine the relative efficiency of the aircraft. Aircraft cost tends to be proportional to empty weight; a simple but reasonably accurate representation of transport cost effectiveness is payload times block speed divided by empty weight. This productivity parameter is plotted in Figure 31 for the three aircraft in question. The airplane has the best payload fraction and highest speed and far outstrips the others at the longer ranges. At short ranges the time spent in traffic patterns and taxiing reduces the block speed of the airplane substantially. The rotary wing vehicles have much smaller unproductive time penalties, and their block speeds exceed that of the airplane to significant ranges. At short ranges, the productivity parameter of the helicopter is superior to that of the airplane, and also to that of the 300-knot rotorcraft for ranges up to more than 100 nm. At long ranges, the 300-knot rotorcraft surpasses the helicopter slightly, but falls far short of the airplane. The helicopter is the aircraft of choice at short ranges whether VTOL capability is required or not. The high speed rotorcraft will be the aircraft of choice only at long ranges when VTOL capability is mandatory, or when the value of speed outweighs the negative economic impact. In any case, the year 2000 is no longer very far away; simple economics will dictate that most rotorcraft in that year will be existing production helicopter types or derivatives of current models.

HEAVY-LIFT HELICOPTERS

Up to this point, only the increased speed potential of rotorcraft has been discussed; the second possible direction of growth is increased size. The maximum gross weight trend of helicopters as a function of time is shown in Figure 32. Two lines are shown: one for Western nations, and one for the USSR, which clearly has a lead in helicopter size over the rest of the world. Their lead in payload capability and productivity is not so striking; Western helicopters tend to have lower empty weight fractions and thus higher payloads for any given gross weight. It is of interest to note that the current largest production Soviet helicopter, the Mil-26, Figure 33, is considerably smaller than its predecessor, the Mil-12, Figure 34, introduced a decade earlier. After experimenting with super large helicopters, the Soviets apparently have concluded that a smaller size vehicle is more practical.

The largest current helicopter in the Western world is the Sikorsky CH-53E, Figure 35, with a rotor diameter of 79 feet (24 m), gross weights up to 73,500 lb. (33,300 kg.) and maximum payloads of 16 tons (14,500 kg). Larger helicopters have been contemplated. The U.S. Army Heavy Lift Helicopter (HLH) program for a substantially larger vehicle was initiated in the 1970's but terminated before an aircraft was completed. At present, the U.S. Army is formulating plans for an Advanced Cargo Aircraft (ACA) which will also be a large helicopter, but with the exact payload and operational requirements still in the process of being defined.

There is an economic problem that must be dealt with when planning a heavy-lift helicopter. If it is sized to carry the largest payload items that the user might ever want to transport by air, then it has to be extremely large. Development costs will be very high, and because of high unit production costs, the number of aircraft that can be afforded will be low, resulting in a high development amortization cost per aircraft which drives unit costs still higher.

Another problem is that, although the theoretical productivity of a fully loaded, very large helicopter can be [illegible] good or better than that of smaller size helicopters, the theoretical productivity is almost never achi[illegible]. Most payloads will be substantially lighter than the maximum, so that the aircraft seldom will be loa[illegible] capacity. Sikorsky Aircraft is currently studying the spectrum of probable payloads to be carri[illegible] large military helicopters. One result is shown in Figure 36, which compares theoretical and avera[illegible]ual productivity (defined somewhat differently than in the earlier example) as a function of design [illegible]ad. The highest achieved productivity is obtained for an aircraft having a design payload consid[illegible] smaller than for the maximum of the study; the smaller aircraft is twice as cost-effective as the larger one. Unfortunately, the smaller aircraft cannot pick up the very large loads occasionally required.

The solu[illegible]n to this dilemma is to use two (or more) helicopters rather than one to pick up the occasional [illegible] heavy load. Just as two or more persons frequently cooperate to carry something too heavy for one, helicopters can also cooperate in a similar fashion. The use of helicopter twin lift is quite an old idea and has had occasional but never routine use. Figure 37 shows a demonstration experiment conducted [illegible] Sikorsky in 1970, where two CH-54 Skycrane helicopters picked up a load of 20 tons (18,140 kg. [illegible] well beyond the capacity of either helicopter alone. The experiment was very conservative in that the spreader bar, designed to keep the two helicopters well separated, was very long, with the result that the spreader bar weight was relatively heavy and the overall dimensions were large. Each helicopter was independently piloted, with voice communication the only contact between the two. The pilot workload was higher than desirable, and it was concluded that a "master-slave" control system, where the "slave" helicopter is automatically controlled and stabilized by the normal control inputs by the "master" pilot, would be needed in order to make the operation practical.

With modern digital flight control technology, the master-slave concept, including overall system stabilization, should be readily achievable, reducing pilot workload and allowing much more compact dimensions as well. The preferred solution to the heavy lift requirement would use twin lift for that small percentage (one or two percent) of missions where the heaviest loads must be carried, and use each helicopter individually for the much larger number of cases where the load is considerably smaller. Figure 38 illustrates the use of two 19-ton (17,200 kg.) payload helicopters to pick up a 35-ton (31,700 kg.) p[illegible]load. The very large helicopter is inefficient for average sized loads; it will be unaffordable for that reason. A moderately large helicopter, incorporating modern technology to achieve excellent payload to gross weight fraction, is a much better choice. Despite somewhat reduced cruise speeds in the twin lift mode and the logist[illegible]cs of having the spreader bar available where needed, higher achieved productivity and substantially lower costs will result. By the year 2000, the twin-lift technique, which can effectively double the lift potential of any size helicopter, should be in fairly widespread use.

CONCLUSIONS

1) The advantages of low-disk-loading rotorcraft vis-a-vis high-disk-loading VTOL concepts are fundamental; rotorcraft are here to stay.

2) The pure helicopter is also here to stay, because of its unexcelled payload and productivity compared to higher speed rotorcraft types at short ranges. Economic factors will dictate that most rotorcraft in the year 2000 will be existing production helicopter types or derivatives of present models.

3) Many high-speed rotorcraft experiments have been tried in the past, but none of the concepts survived to the production stage. The primary reason for this is believed to be the relatively poor economics of the vehicles investigated, with the then-available technology levels.

4) New technology can be applied to improve the attributes of advanced rotorcraft; some configurations will be much more competitive than they were originally, and new types are made feasible.

5) Configurations believed potentially viable, in order of increasing maximum potential speeds, are the compound helicopter, the ABC, the tilt-rotor, the X-Wing, and the stowed-rotor.

6) The need for very-heavy-lift helicopters can best be satisfied by developing a modern, moderately large helicopter that can lift most but not all of the desired payloads. To lift the relatively rare heaviest loads, a twin-lift technique should be utilized. Substantial efficiency gains and cost savings will result.

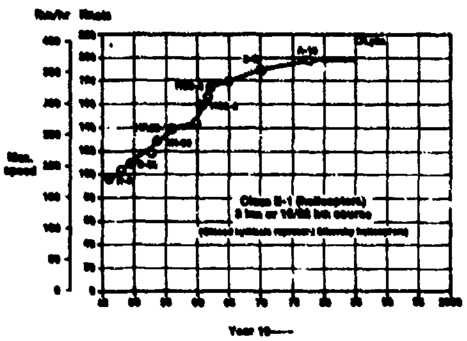

Figure 1. MAXIMUM OFFICIAL SPEED RECORDS

Figure 5. McDONNELL XV-1

Figure 2. SIKORSKY S-76 MARK II HELICOPTER

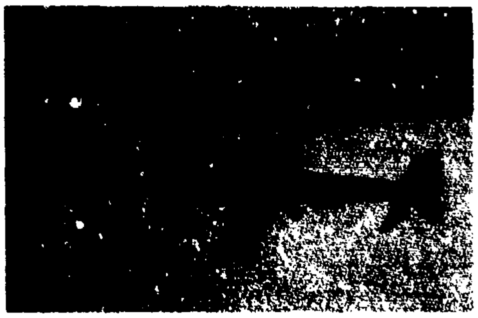

Figure 6. S-67 BLACKHAWK PROTOTYPE GUNSHIP

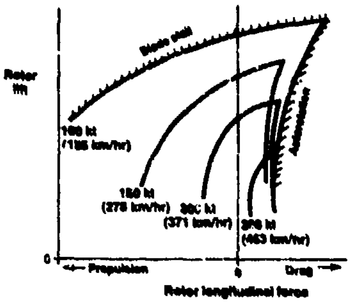

Figure 3. EFFECT OF SPEED ON ROTOR FORCE CAPABILITIES

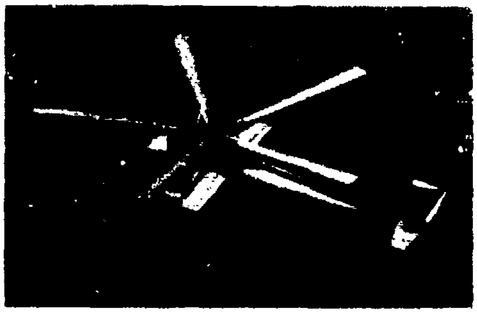

Figure 7. NH-3A (S-61F) RESEARCH COMPOUND HELICOPTER

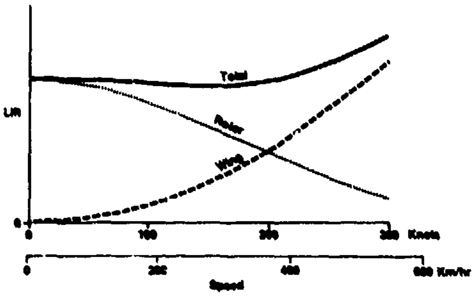

Figure 4. COMBINED LIFT OF ROTOR AND WING

Figure 8. BELL UH-1 COMPOUND

Figure 9. FAIRRY ROTODYNE

Figure 13. SIKORSKY TILT ROTOR DESIGN

Figure 10. LOCKHEED AH-56 CHEYENNE

Figure 14. FOLDING TILT ROTOR DESIGN

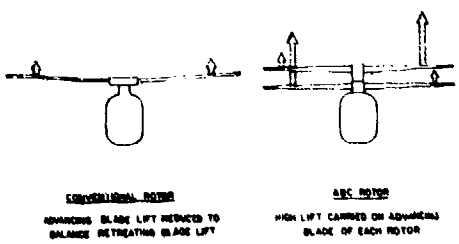

Figure 11. BASIC PRINCIPLE OF ADVANCING BLADE CONCEPT

Figure 15. BELL XV-3

Figure 12. XH-59A DEMONSTRATION AIRCRAFT

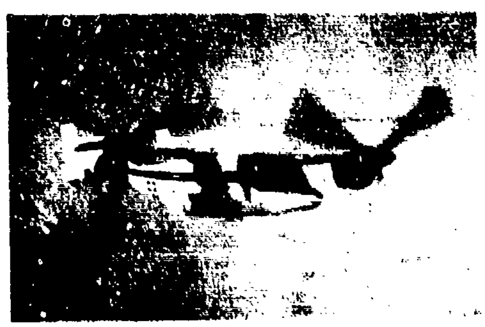

Figure 16. BELL XV-15

Figure 17. X-WING ROTOR ON NASA ROTOR SYSTEMS RESEARCH AIRCRAFT

Figure 18. NAVY X-WING CONCEPT

Figure 19. X-WING AIRFOIL WITH CIRCULATION CONTROL

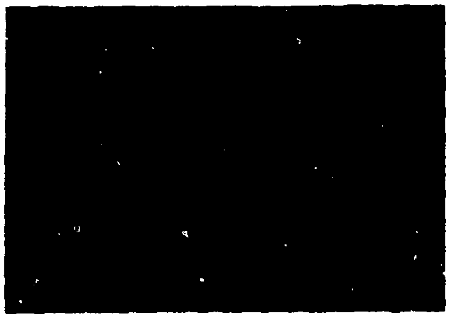

Figure 20. X-WING ROTOR BLADE

Figure 21. X-WING AIR SUPPLY VALVES AND DISTRIBUTION DUCTS

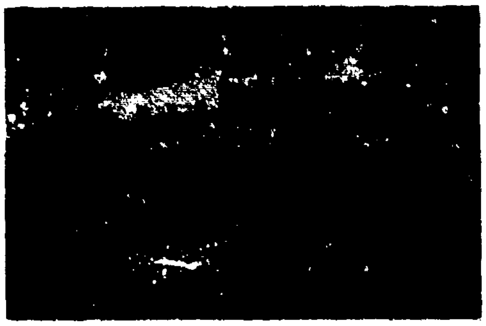

Figure 22. SIKORSKY STOWED ROTOR AIRCRAFT CONCEPT

Figure 23. LOCKHEED STOWED ROTOR CONCEPT

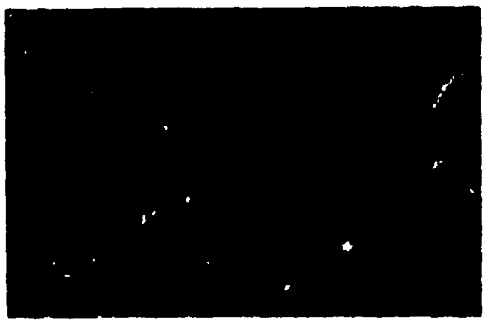

Figure 24. VARIABLE DIAMETER STOWED ROTOR CONCEPT

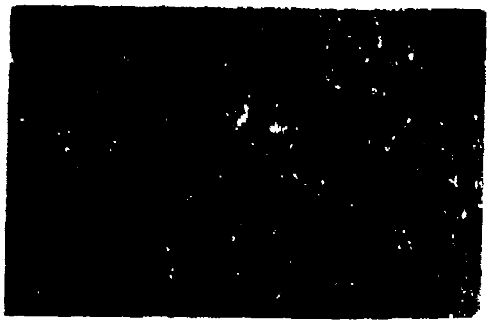

Figure 25. VARIABLE DIAMETER ROTOR WIND TUNNEL TEST

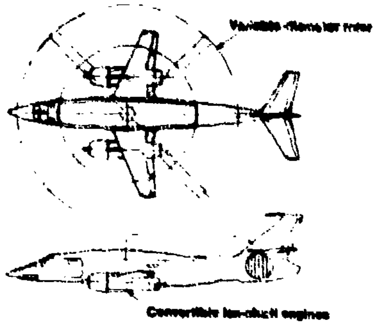

Figure 29. HIGH SPEED STOWED ROTOR DESIGN

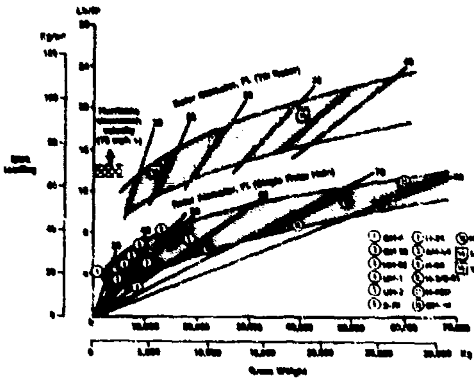

Figure 26. DISK LOADING TRENDS

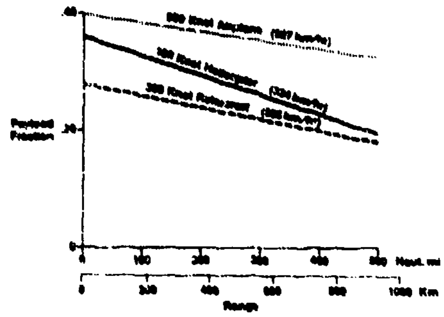

Figure 30. COMPARATIVE PAYLOAD FRACTIONS

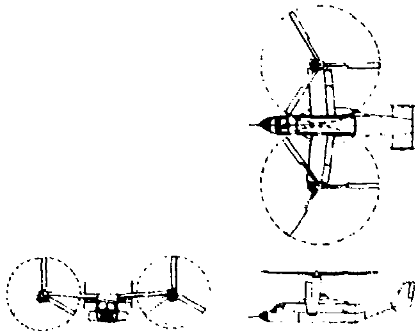

Figure 27. VARIABLE DIAMETER TILT ROTOR AIRCRAFT

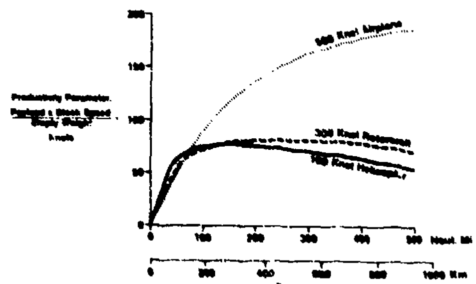

Figure 31. COMPARATIVE PRODUCTIVITY

Figure 28. ADVANCED X-WING AIRCRAFT

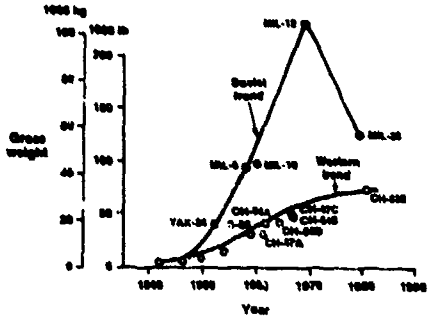

Figure 32. MAXIMUM GROSS WEIGHT TRENDS

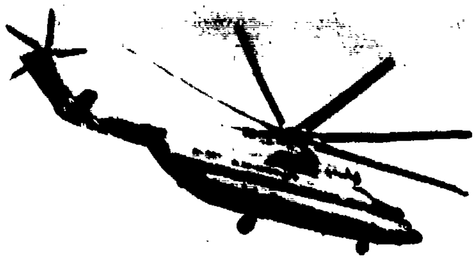

Figure 33. MIL-26 HELICOPTER

Figure 34. MIL-12 HELICOPTER

Figure 35. SIKORSKY CH-53E SUPER STALLION

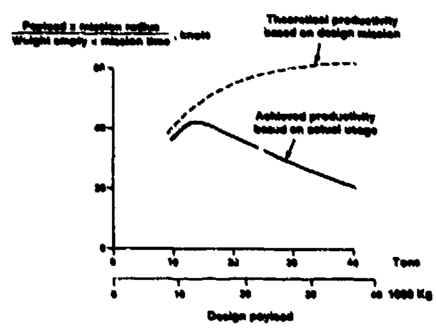

Figure 36. EFFECTS OF DESIGN PAYLOAD ON PRODUCTIVITY

Figure 37. TWIN-LIFT DEMONSTRATION WITH CH-54's

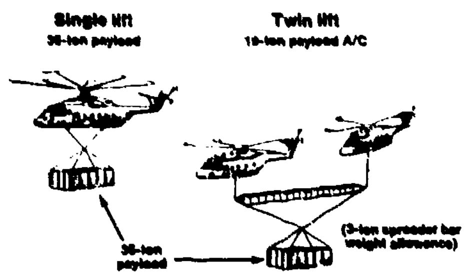

Figure 38. SMALLER AIRCRAFT PERMITTED BY TWIN LIFT

HELICOPTER (PERFORMANCE) MANAGEMENT

by

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SUMMARY

The next generation of military helicopters have to fly under night and adverse weather conditions below top tree level. Dornier System started 15 years ago to develop avionic systems with a performance which allow the pilots to fly the new missions. In the Dornier baseline display cockpit concept the first steps of a Helicopter Management System (HMS) will be realized. This paper will discuss the main aspects of the system. Its main parts are the Dornier AFA mission planning system on the ground and the HMS onboard the helicopter. The basic functions of the system will be explained.

The main point, is to use the same performance data base on the ground as in the helicopter. This data will be the performance data set for the helicopter Flight Manuals.

1. INTRODUCTION

The electronic 'glass' cockpit has proved its worth in the past years and there have been significant improvements in reliability, maintainability and aircraft availability with cathode ray tubes, CRT's. Performance data computers were developed to look ahead and optimize fuel use and trajectories to guide the aircraft. In the late 1970's the development of digital Flight Management systems was initiated for various aircraft. It is possible now to establish the onboard computer capacity that is necessary for flight and performance management systems and the considerable onboard data bases that could aid the crew in mission conduct, tactics and vehicle system management. But advances should also be made on the sensor side, for example with wind sensors and weight sensors in the landing gear for load control.

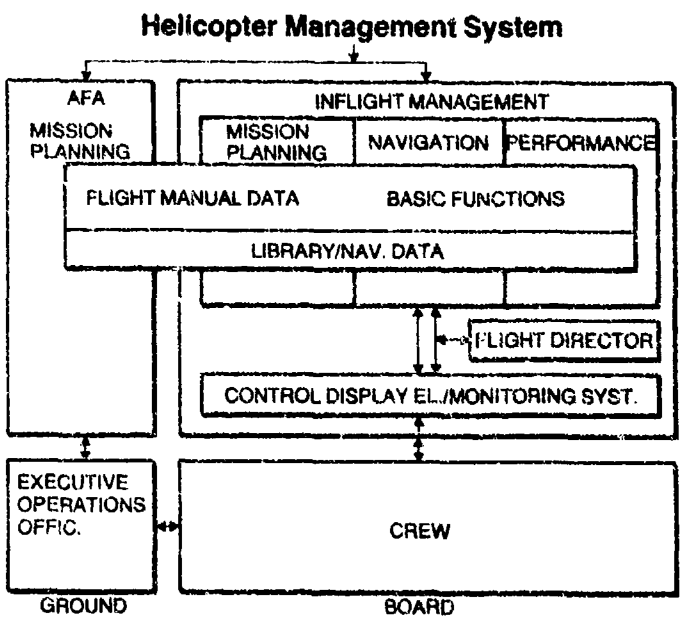
Dornier's flight tests [1] cover Fly By Wire control, Pilot Night Vision and Integrated Avionics. In the NSC Program (Nachtsichtcockpit, Night Vision Cockpit) the main points of investigation besides Night Vision are

- Controls and CRT Displays (symbology, optical and acoustical warnings)
- Navigation (integration of map displays e.g.).

2. CONCEPT OF A HELICOPTER MANAGEMENT SYSTEM (HMS)

A Helicopter Management System, HMS, has been conceived utilising Dornier ground mission planning activities, advanced cockpit studies and night vision flight tests.

Fig. 1



The ground based part of the system is used to conduct mission planning in a very quick and convenient way (AFA). The onboard system part is to be used for mission planning and also for special replanning, performance management and aircraft monitoring. Appropriate new sensors are needed.
Fig. 1 shows the concept in general.
The HMS shall rationalize, automize and optimize planning and dispositioning on the ground and inflight. It shall reduce workload too. Helpful to this is an information and monitoring concept based on the cockpit information requirements.

Fig. 2 **Overall Development Aims**

	GROUND	HELIC.
● USE THE PERFORMANCE POTENTIAL	●	●
● COVER NEW TASKS (NIGHT, WEATHER, CONFLICT)	●	●
● INCREASE FLIGHT SAFETY	●	●
● MINIMIZE REACTION TIME	●	
● REDUCE PILOT WORKLOAD		●
● ESTABLISH FLEXIBILITY		●
● SAVE MONEY		●

3. SYSTEM FUNCTIONS, DATA BASE

Main functions and data base of the AFA, the ground based part of the system

The helicopter computerized mission planning system AFA calculates the mission feasibility based on the assumed flight.
With stored information on the tactical threat situation and the required mission or task

- the flight track is calculated automatically and very quickly. The AFA hardware comprises a
  - o high resolution colour raster display for the dialogue and graphic display of the planning results
  - o video hardcopy unit for documentation of the planning results
  - o map digitizer for the simple extraction of co-ordinates from maps of different scales and for the provision of a menue field
  - o computer with mass storage, to carry out flight track, time and fuel, and navigational calculations for various mission types
  - o printer for alphanumeric output and a
  - o mission data output device

Furthermore

- coordination of combined operation in the target area is possible
- optimization of the fuel consumption in peace time can be done
- the mission data output device for the electronic storage of the planning results on a mission memory device (MDT) can be used to store the data in the aircraft.

The so-called long-lead data comprises the following,

- weapons system data consisting of:
  - o description of the aircraft configuration (conventional loads)
  - o description of the engine configuration
  - o weapon data base, etc.
  - o airfields: storage of the main airfield data which can be called up during the planning for take-off; landing, or emergency landing
  - o magnetic variation data (declination) for calculating the magnetic course correction
  - o point data: storage of data describing,among others, TACAN stations, beacons, contact points, and several thousand geographical points. These points can be called up in the planning as navigational fix points.
  - o routes: storage of route segments which can be called up as parts of the total flight track.
  - o map data: description of any kind of maps having the scale of 1:50,000, 1:100,000, 1:250,000, 1:500,000 and 1:1,000,000. For the planning, these data can be used for adjusting the desired planning map on the map digitizer.

- AFA data comprising
  - o equipment configurations
  - o software configurations

HMS airborne, functions, data base

The Management System onboard the helicopter must reduce the workload of the crew, help in the planning and, forecasting and it must optimize, inform and monitor.
Five main tasks of the HMS can be identified.

To fulfill these tasks basic performance calculations are necessary to obtain the flight track or flight path, flight durations and fuel consumption etc.

Fig. 3

HMS - Tasks

1. PLAN AHEAD
   - MISSION PLANNING
   - INFLIGHT (RE)PLANNING
   - FEASIBILITY, CRITICAL CASES
2. ACTION INSTRUCTIONS, CHECKLISTS
   - MISSION MODE CHANGE (AUTOM.)
   - ON DEMAND
   - BY WARNING & CAUTION SITUATIONS
3. MONITORING FLIGHT / MISSION STATUS
4. ADDIT. INFORMATIONS E.G.
   - MAX. TORQUE AVAILABLE
   - MAX. CLIMBING SPEED
   - MAX. DESCENDING SPEED
5. FLIGHT PATH CALCUL FOR FLIGHT DIRECTOR

Data model

The following Libraries are necessary for this,

- Performance Data and
- Navigation Data, point data, e.g.
  - landing areas with
    - fuel and ammunition supply
    - navigational aids
    - landing aids
    - repair facilities
  - hospitals, actual threat areas
  - restricted areas.

An adapted sensor concept must be foreseen to get the right system inputs with a suitable accuracy.

The main point in the proposed HMS concept is to have an identical helicopter performance data set in the AFA mission planning system on the ground and in the helicopter. As far as mission planning and replanning is concerned the flight manual performance data base is to be used for this purpose.

The graphs in Fig. 4 show in principle how data is made available. The example (UH-1D) gives fuel flow against altitude and A/C velocity and Gross Mass (GM). It also shows power against velocity and gross mass, and range and endurance performance curves.

The following table gives a short comparison of the main data and planning features if the HMS data base is organized in this way.

Fig. 4 PERFORMANCE DATA

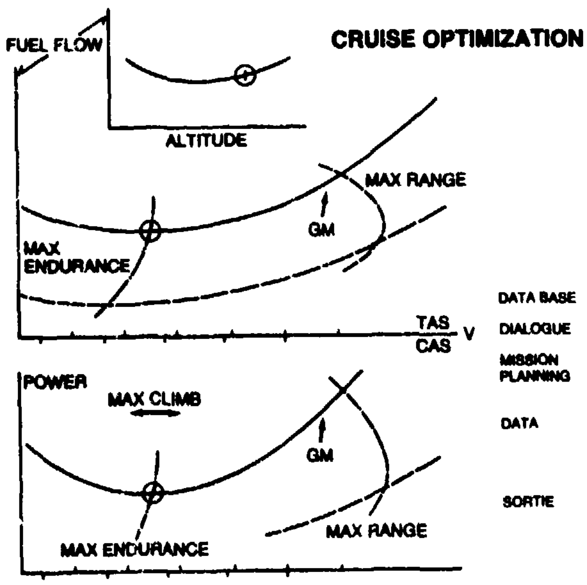


Fig. 5 DATA BASE

	Helicopter Management System: GROUND	ABOARD
DATA BASE	BROAD	SMALLER LONG LEAD
DIALOGUE	SIMPLE, QUICK	LIMITED CAPABILITY
MISSION PLANNING	SIMPLE, CAREFUL	REPLANNING CAPABILITY DETOURS, TARGET CHANGE
DATA	PLAN DATA ON TACT. SITUATION WEATHER, FUEL	ACTUAL DATA ON TACT. SIT., WEATHER, FLIGHT DURATION PERFORMANCE, FUEL
SORTIE	TIME SYNCHRONIZED PLANNING OF SEVERAL A/C	

Onboard system realization

The onboard system functions are shown by fig. 6.7.

Fig. 6 **System - Parts**

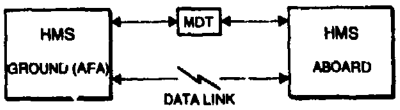
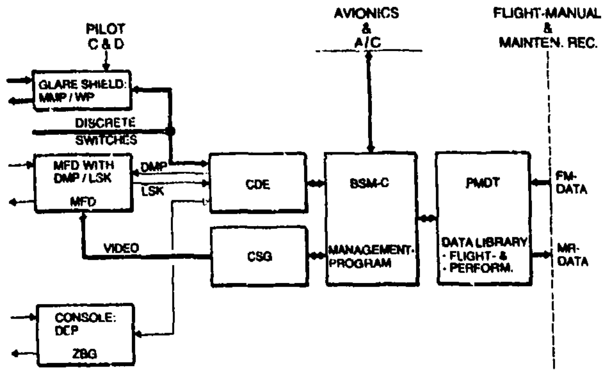


Fig. 7 **HMS in the Integrated System**



The hardware realization is given by figure 7. The test vehicle configuration has been described in [1].

- Interface ground/board

The HMS needs

- AFA mission plan data, loaded by a MDT, combined with threat situation data if possible (actual data)
- navigational points libraries (long lead and actual data)
- performance data (long lead data)

- data storage

Actual data (MDT) and long lead data must be stored in a proper way so that data change is feasible

- Management part

This part of the HMS will be located in the Bus System Mission Controller or in a separate box.

- Controls and Displays, C + D

Interaction and communication between the operator and the HMS is to be done by the data input devices and the displays, cathode ray tubes CRT's. All displays are multifunction displays (fig. 14), MDF's.

The display cockpit concept is described in 6.

4. TASKS

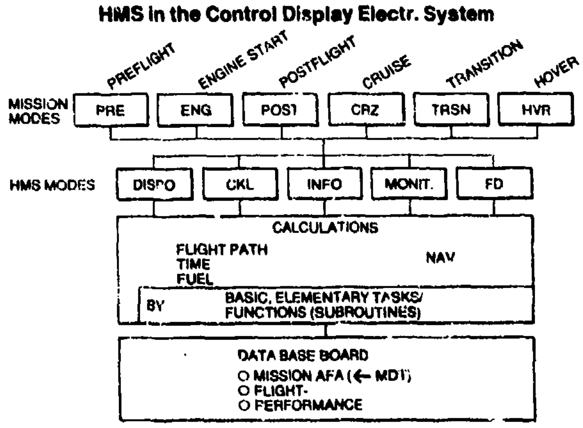
Mission Modes, Flight phase, tasks

The HMS system modes: DISPOSITION, CHECKLISTS must be usable in all flight phases. These phases are

- PREFLIGHT	PREF
- ENGINE-START	ENG
- POSTFLIGHT	POST
- CRUISE	CRZE
- TRANSITION	TRSN
- (extended) HOVER	HVR

Fig. 8

HMS in the Control Display Electr. System



The main tasks which have to be done in the HMS-Mode are the calculations which are neccessary to determine

- flight track
- flight durations (between way points etc.)
- fuel
- mission performance capabilities

as listed in the following.

Flight path calculations

In general

- planned, acutal position known
- electronic chart necessary
- points library necessary
- good navigational aids

Requirements:

- flight path (→ route planning)
- flight time (→ time planning, reserves)
- flight part requirements
  - position actual/planned
  - destination
  - flight path, way points, bases, intermediate targets
  - restricted areas
  - hospitals
  - refueling points
  - airfields with precision approach facilities
- flight time requirements
  - time for mission and/or to several waypoints at V, V_{max}, V_{max} range, V_{max} endurance
  - airspeed required depending on time planning
  - times of arrival possible
  - time over target (≤ 30 sec window) possible
  - time over several waypoints
  - flight over sea: point of return
  - time on scene, loiter time on planned routes
  - time for detour
- fuel requirements
  - how far/for what time sufficient fuel at V, V_{max} ect.
  - fuel enough for planned route/plus flight to refueling point with reserves?
  - reserves
  - alternatives to destination
  - alternative altitude, climb rate, fuel consumption, wind velocities
  - power setting, RPM, Gross Weight
- optimal range, endurance

Fuel-/Range Optimization is possible. See fig. 4

As a case to illustrate the HMS process in its running, some aspects of a plan ahead are described for (extended) hover (e.g. hot and high).

The case could be:

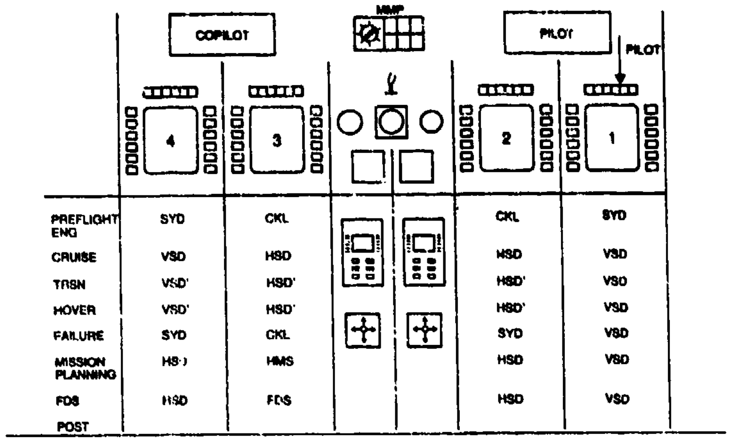
- replanning or actual inflight dispositioning i.e. plan ahead
- flight case to calculate: transition, hover hot and high or
- hoist operation in hover

Fig. 9 shows the interdependence of mission mode and actual display mode. The following is an example of inflight operations.

- In the present mission mode CRUISE the
- copilot assigns display mode HMS to MFD 3 (by the Display Mode Panel of MFD 3) and of
- Mission Mode Hover by the Mission Mode Panel (MMP).

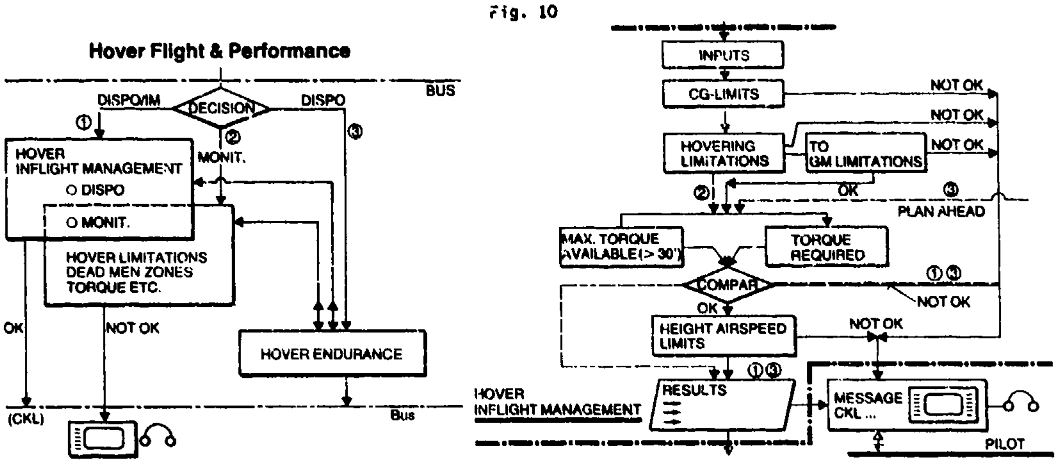
Now a hover mission part can be precalculated or preplanned during the Cruise Mode by using MFD No. 3. Data inputs can be made by the Line Select Keys (LSK's), the Data Entry Panels (DEP's) and/or the Central Data Entry Panel (ZBG).

Fig. 9 DISPLAY IMAGES MATRIX



Disposition by the basic function "Hover" could be done as the following figures show:

Fig. 10



- Processing of pilot and sensor data inputs and actual system values and results (GM, CG-position, fuel)
- Calculation of hover conditions, with inputs of the

  - probable wind velocity
  - planned or estimated hover time
  - altitude
  - external load

  can be calculated, if
  - hover limits can be held
  - CG-limits are held
  - there is enough torque
  - dead man's regions are affected.

- Results are

  - calculated values if limits are not exceeded
  - messages/warnings if limits not held, output of the results as checklist or values in the VSD, HSD.

<u>Overall top down structure of the Helicopter Management System</u>

This complex system Helicopter Management can be structured

- into its five main tasks as shown
- during all mission modes, and
- its necessary basic tasks like the calculation of "Hover Endurance" and elementary functions such as the computation of "Density Altitude".

Figure 11 shows this in a top down structure while fig. 12 lists most of the necessary basic and elementary functions.

Fig. 11 TASKS

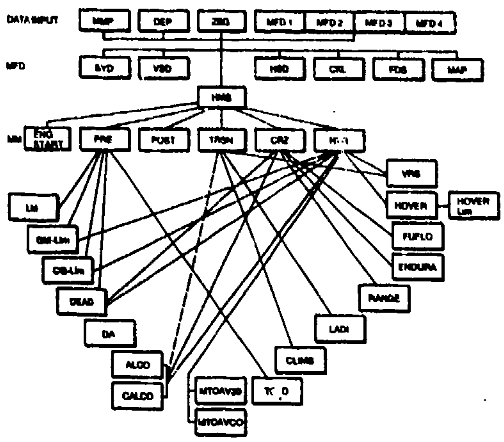


Fig. 12 "SUBROUTINES" TO CALCULATE PERFORMANCE (UH-1D)

LM	Load control
GM-Lim	Gross mass limitations
CG-Lim	Center of gravity lim.
Dead	Dead men's zones
DA	Density altitude
ALCO	Altitude Correction
CALCO	Calibrated airspeed correction
MTOAVCO	Max. Torque available
MTOAV30	Max. Torque available, 30' Limit.
TOLO	Take off, Lift off
LADI	Landing distance
ENDURA	Endurance
FUFLO	Fuel flow
VRS	Vortex ring state

All these functions and data set are

- structured obligatory as the GAF T.O. (GERMAN AIRFORCE TECHNICAL ORDER) of Helicopter Flight manuals requires, they can be modified and interpolated as standard require
- usable in any order as often as necessary with standardized data exchange and similar data bus interface to store results of planning and calculations if neccessary and possible
- bound into the monitoring concept according to the flown MODE, as the GM/CG-Limitations, height airspeed limits, torque limits, wind directions, wind limits, hoist operation limits etc.

5. DATA AND SENSOR ACCURACY

For the airborne part of a Helicopter (Performance) Management System appropriate cockpit and data requirements are necessary. To establish task flexibility by using all the performance potential of the aircraft all calculations must be done with a proper accuracy.

Detailed simulation and error models have to be developed for all the relevant sensor equipment and environmental disturbances as a primary design analysis and performance assessment tool.

Fig. 13 shows some brief results from an investigation into weight sensor accuracy for load control.

Fig. 13 LOAD CONTROL ACCURACY

GM-/ CG-SENSORS / LOAD CONTROL

- WEIGHT-SENSORS IN THE LANDING GEAR NECESSARY
- GM AND CG-LAT./LONG. MUST BE CALCULABLE, ALSO ON SLOPES
- RELEVANT ONLY ERRORS IN LAT/LONG-DIRECTION (→ CG-LIMITS)

CG-ERRORS LAT./LONG.

- DO NOT DEPEND ON WEIGHT & MASS-DISTRIBUTION,
- NEARLY NOT ON ATTITUDE-ACCURACY
- DEPEND ON WHEEL-BASE
- WILL BE IN THE MAGNITUDE OF cm
- SENSOR ACCURACY/ANALYSIS OF ERROR MARGINS, TOLERANCES IS NECESSARY

6. COCKPIT CONCEPT

The operation philosophy of the Dornier HMS is based on using all the possibilities and advantages of a modern electronic and display cockpit concept. The requirement for this, is the adaptation of all usable techniques to the human operator to reduce pilot workload. The interdependencies to be accounted for are

- mission requirements
- degree of automatization, pilot work load
- cockpit layout, interface cockpit-operator

Fig. 14 ELEMENTS OF MODERN COCKPIT DESIGN

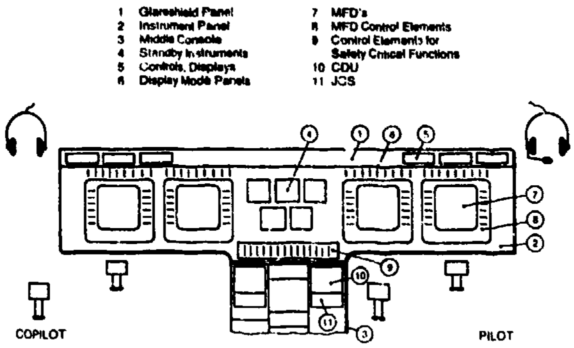


Fig. 15 MISSION MODE PANEL

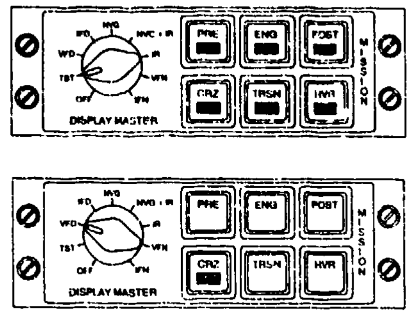


Fig. 14 shows a preliminary overall view of a possible cockpit configuration. Main instruments for the concept are

- MFD's, Multi Function Displays with the possibility to assign each display format for any mission mode
- MMP's, mission mode panels by which the images of any mission mode can be assigned to any one of the several MFD's, Fig. 15
- DEP's, data entry panels for the input of alphanumeric data to initiate HMS calculations.

Monitoring of errors, malfunctions etc.

The HMS must check if

- pilot inputs
- calculation results
- sensor data

are plausible. Cross check capabilities must be provided and utilized, as for example comparison of several sources of fuel data (fuel flow measured and fuel remaining, calculated fuel consumption and fuel remaining). The HMS must check if limits are exceeded and the monitoring concept must give a notice or a warning/caution advice in the MFD-mode.

The Dornier electronic cockpit flight test background (Night Vision Cockpit) has been described in detail in Ref. [1].

7. CONCLUSIONS

The Dornier helicopter performance management preliminary design gives some advantages that are listed in the following table:

- Know-how transfer from α-Jet/Tornado AFA's to helicopters
- Same data base ground/airborne, results comparable
- Same data set structure for several helicopters reduces development time

- AFA mission planning system could be standardized for all
  - forces and A/C types
  - rotor and fixed wing A/C

- Helicopter Management System furthermore
  - reduces pilot workload
  - reduces reaction time
  - makes new tasks possible
  - contributes to flight safety
  - exploits performance potential
  - establishes flexibility

Cost effectiveness of a joint board/ground data base and calculation programs will be given similarly.

But there is still a lot of work to be done. One of the major problems is to specify the necessary sensor accuracies, to define data requirements and to prove these e.g. by simulation.

Nomenclature

AFA	Mission Planning System
BSM-C	Bus System Mission Controller
CDE	Control Display Electronics
CKL	Checklist
CRT	Cathode Ray Tube
DEP	Data Entry Panel
DMP	Display Mode Panel
FDS	Flight Director System
GM	Gross Mass
HMS	Helicopter Management System
HSD	Horizontal Situation Display
LSK	Line Select Key
MAP	Map
MD1	Mission Memory Device
MMP	Mission Mode Panel
MR	Maintenance Recorder
PMDT	Long Lead Data Storage
To	Take Off
VSD	Vertical Situation Display
WP	Way Point
ZBG	Central Data Entry Panel

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CONDUITE DE TIR HELICOPTERE
INTERETS D'UNE POURSUITE AUTOMATIQUE DE CIBLE

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RESUME

Cet exposé montre les divers intérets que présente l'implantation d'une poursuite de cibles dans le cadre d'une conduite tir hélicoptère.

L'exposé, après une description succincte des principaux éléments intervenants dans une conduite de tir (viseurs, armements, calculateurs), de leurs rôles respectifs dans le déroulement de la prise à partie d'une cible, détaille l'intérêt d'une poursuite automatique de cibles à traitements d'images en termes :

- de charge de travail de l'opérateur,
- d'amélioration du trainage et des bruits de poursuite,
- d'amélioration des informations de cinématique (précision, bruits) de la cible fournies à la conduite de tir.

L'exposé présente enfin les divers critères retenus pour la conception de la commande.

1 - DESCRIPTION GENERALE

1.1. TYPES DE CIBLES

Les types de cible qu'une conduite de tir hélicoptère peut prendre à partie sont très divers. On peut citer :

- objectifs de jour ou de nuit,
- objectifs terrestres (air-sol),
- aéronefs évoluant dans l'espace aérien proche du sol (air-air).

Pour remplir ces missions, l'hélicoptère est équipé, en plus de son système d'armes, d'un ensemble de systèmes optiques permettant l'acquisition des cibles et leur désignation aux systèmes d'armements. Le système optique est donc le périphérique principal de la conduite de tir.

1.2. PRINCIPAUX CONSTITUANTS D'UNE CONDUITE DE TIR HELICOPTERE

1.2.1. Les viseurs (le système optique)

Leur rôle est le suivant :

- observation et détection de cibles,
- reconnaissance et identification de cibles,
- désignation de cibles au profit d'un autre viseur ou d'une arme.

Certains types de viseurs ont également le rôle d'être :

- apte à être un capteur d'image à fort grossissement et stabilisées dans le domaine visible et infrarouge,

- apte à être un organe de visée de grande précision et de mesure de l'éloignement de la cible (mesure des paramètres cible : angles, vitesse et distance).

1.2.1.1. Le viseur dit "TIREUR"

Il est à la disposition du copilote. Parmi les montages possibles sur hélicoptère, on peut citer le montage de toit, le montage de mât, le montage de nez. La figure suivante présente un viseur de toit.

VISEUR TIREUR

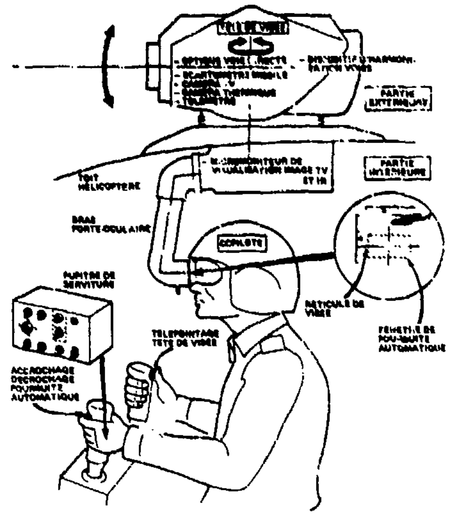


FIGURE 1

Il se compose des deux parties suivantes :

- une partie extérieure à l'hélicoptère, constituée d'une plateforme gyrostabilisée orientable en site et en gisement et renfermant les différents senseurs nécessaires aux fonctions :

  * d'observation et d'identification de cibles (voie optique directe, caméra TV, caméra thermique),
  * de désignation et d'engagement de la poursuite automatique (caméra TV, caméra thermique),
  * de mesure de distance de la cible (télémètre laser),
  * d'harmonisation des différentes voies d'observation.

- d'une partie intérieure à l'hélicoptère, comprenant un système de présentation d'images permettant la visualisation à l'opérateur de différentes voies (voie directe optique, caméra TV, caméra thermique).

Le copilote possède également un certain nombre de commandes de mise en oeuvre du viseur disposées sur des manches ergonomiques et sur un pupitre de servitude annexe permettant l'emploi de toutes les fonctions du viseur et notamment l'engagement de la poursuite automatique.

1.2.1.2. Le viseur dit "PILOTE"

C'est un viseur tête haute. Il est fixé à la structure sans partie mobile. Son principe de fonctionnement est tel qu'il permet de superposer devant l'oeil du pilote, une symbologie de pilotage et de désignation de cibles propre au tir sur le paysage extérieur.

Il est composé essentiellement d'une tête de visée permettant la génération de symbologie et de réticules collimatés à l'infini.

1.2.1.3. Le viseur dit "DE CASQUE"

Le pilote et le copilote peuvent disposer d'un viseur de casque. Il se compose de deux parties essentielles :

- un dispositif optique permettant la présentation d'un réticule de visée devant l'oeil du pilote ou du copilote,
- un système d'émetteurs et de capteurs permettant la reconstitution de la position de la ligne de visée.

1.2.2. Les armements

Les armements sont adaptés à la distance d'engagement et affectés au tir air-air ou air-sol.

- Canon : l'utilisation du canon est réservée à l'engagement à courte et moyenne distance (< 1500 m) contre des objectifs air-air.
- Missiles : l'utilisation des missiles est réservée à l'engagement à moyenne et longue distance (< 4000 m) contre des objectifs air-air.
- Roquettes : l'utilisation des roquettes est réservée à l'engagement à moyenne et longue distance (< 4000 m) contre des objectifs air-sol.

1.2.3. Les calculateurs d'armements

Leur rôle est d'assurer la gestion de l'ensemble du système de conduite de tir. Ils assurent la gestion des modes systèmes, l'attribution des diverses ressources (viseurs, armes) au pilote et copilote et les calculs de corrections de tirs nécessaires à certains armements.

2 - MÉTHODE D'ACQUISITION D'UNE CIBLE

L'utilisation des viseurs est différente suivant leur type. L'ergonomie et les possibilités de chaque type de viseur sont adaptées aux possibilités de l'utilisateur (charges de travail différentes) et aux types de tir envisagés.

Viseur tireur : - moyen d'acquisition, de désignation et de poursuite de cibles pour des tirs préparés et les tirs à longue distance.
- pilotage de nuit en secours.
- recalage de la navigation.

Viseur pilote : - instrument principal de pilotage machine.
- moyen d'acquisition et de désignation de cibles pour des tirs axiaux des différents armements.

Viseur de casque : - moyen d'acquisition et de désignation de cibles au profit du viseur tireur.
- moyen de pointage et de tir de riposte à courte distance.

L'acquisition d'une cible est réalisée par un des viseurs du système d'arme par pointage d'un réticule représentatif de la ligne de visée sur l'objectif.

Le pointage d'une ligne de visée peut être réalisé en général de deux manières :

- soit le réticule de visée est pointé sur la cible par un opérateur grâce à un manche,
- soit le réticule de visée est pointé sur la cible à partir d'un asservissement sur des consignes de désignation de cibles fournies par un autre viseur.

La figure suivante illustre la logique d'acquisition d'une cible par le viseur tireur.

LOGIQUE D'ACQUISITION ET DE POURSUITE
MANUELLE D'UNE CIBLE PAR LE VISEUR TIREUR

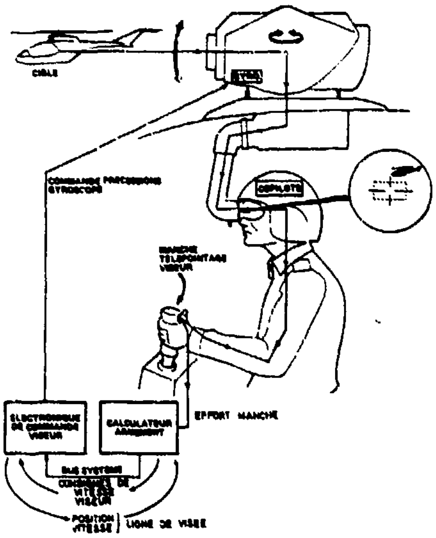


FIGURE 2

La méthode utilisée ici est un pointage manuel de la ligne de visée (réticule) sur la cible.

La figure suivante présente la logique de fonctionnement du viseur en phase de poursuite automatique.

LOGIQUE DE POURSUITE AUTOMATIQUE D'UNE CIBLE PAR LE VISEUR TIREUR

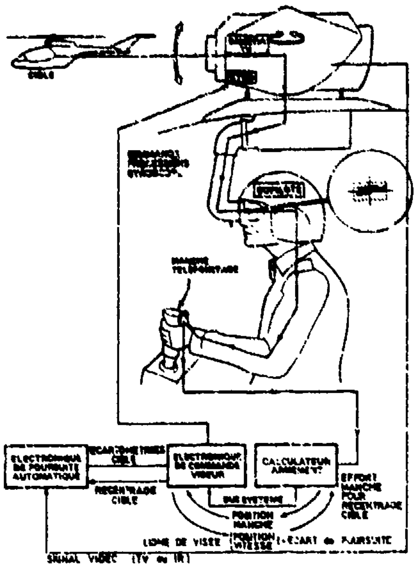


FIGURE 3

Cette phase est la phase logique succédant au pointage manuel de la cible et démarre avec l'engagement de la poursuite automatique par un organe de commande.

Le signal vidéo (TV ou IR) est exploité par une électronique de poursuite à traitement d'images. Cette électronique délivre des écartométries de position de la cible dans le champ sur lequel s'effectue la poursuite à l'électronique de commande viseur qui effectue alors l'asservissement de la ligne de visée sur la cible.

3 - POURSUITE AUTOMATIQUE

3.1. INTERETS D'UNE POURSUITE AUTOMATIQUE

Les avantages d'une poursuite automatique par rapport à une poursuite manuelle sont les suivants :

- charge de travail du tireur : en diminution par rapport à une poursuite manuelle. Seules restent à la charge du tireur la surveillance du suivi et l'opération de recentrage de la cible.

- bruits de visée : il s'agit de l'écart entre la cible et le réticule de visée (trainage). Cet écart est diminué d'un rapport 5 à 10 en poursuite automatique.

- précision de la position de la ligne de visée : en poursuite manuelle, la position de la ligne de visée fournie à la conduite de tir est la position du réticule de visée.
En poursuite automatique, cette position est corrigée des écartométries de poursuite. Elle représente donc la position de la cible, information nécessaire à la conduite de tir.

- qualité de l'information vitesse de la ligne de visée : le bruit superposé à l'information de vitesse est en forte diminution en poursuite automatique. Les calculs de prédiction effectués par la conduite de tir pour l'anticipation de position du canon sont donc beaucoup plus précis.

3.2. PRINCIPAUX CRITERES DE CONCEPTION - CONTRAINTES

Le but à atteindre est d'élaborer une méthode de calcul permettant de réaliser un bouclage optimal au sens d'un critère fixé (temps de réponse, forme de la réponse, minimum d'énergie, ...) d'un viseur gyrostabilisé à une électronique de poursuite automatique.
Les contraintes à prendre en compte sont de deux types : contraintes sur les modèles et contraintes sur la commande.

- Contraintes sur les modèles :

* le modèle du viseur varie en fonction de la situation de vol, de l'angle de visée (perturbations, variations d'inertie,...).
* l'électronique de poursuite peut délivrer des écartométries perturbées lorsque le contraste cible/fond devient trop faible ou lorsqu'il varie trop brusquement (passage de ligne d'horizon, passage sur fonds perturbés, passage de masques totaux ou partiels).
* la fonction de transfert de l'électronique de poursuite est non linéaire (retard pur).

- **Contraintes sur la commande** :

  * bonne fiabilité de la boucle (tenue aux écartométries perturbées).
  * bonne robustesse vis à vis des variations de modèles.
  * bonne robustesse vis à vis de la précision des coefficients du correcteur.
  * bonne facilité d'intégration (nombre de paramètres de réglages minimal).
  * consommation d'énergie minimale.
  * adaptation aux systèmes comportant des retards purs.
  * volume de calcul le plus faible possible.
  * pas de dégradation notable de l'observation et de la prise d'image.
  * trainage minimal compatible de la divergence de la télémétrie.

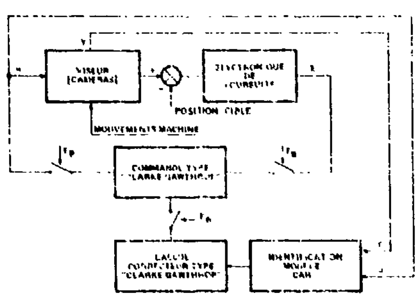
3.3. CHOIX ET PERFORMANCES

La solution aux contraintes évoquées précédemment est une structure auto-adaptative permettant d'intégrer les contraintes de variations de modèles.

La commande est une commande prédictive du type "CLARKE-GAWTHROP" permettant la résolution de toutes les contraintes présentées précédemment, sa supériorité par rapport à des commandes plus classiques résidant principalement dans sa robustesse, sa facilité de réglage (1 paramètre) et sa conception de base adaptée aux systèmes à retards purs.

La figure suivante présente un synoptique de l'asservissement.

SYNOPTIQUE BOUCLAGE AUTOADAPTIF



COMMANDE PRECESSION VISEUR
POSITION LIGNE DE VISEE (AXES MACHINE)
ECARTOMETRIES CIBLE
FREQUENCE DE BOUCLAGE
FREQUENCE D'AUTOADAPTATION
CAR AUTO REGRESSIF A COMMANDE

FIGURE 4

Il consiste en une double boucle :

- une boucle de commande à la fréquence d'asservissement (50 à 100 Hz traditionnellement).
- une boucle d'adaptation du correcteur à une fréquence plus basse palliant les variations lentes de modèles (quelques Hz).

Cette structure à deux étages est simplifiable si les variations de modèle restent faibles.
Il y a lieu de préciser cependant que le volume de calcul nécessaire à ce type de commande peut être qualifié de "assez élevé" (polynomes d'ordre 7 pour les correcteurs) compte-tenu des puissances de calcul embarquables et de "très élevé" si il y a nécessité de la présence de la boucle d'auto-adaptation du correcteur.

3.4. PRESENTATION SUCCINCTE DE LA COMMANDE

La commande de CLARKE-GAWTHROP est une commande à minimisation d'un critère quadratique à un pas.
Elle minimise le critère suivant

$$J = \mathcal{E}(\| \phi(t + k + 1) \|^2)$$

avec $$\phi(t + k + 1) = A^M y(t + k + 1) + \Lambda u(t) - \gamma B^M y_r(t)$$

$y(t + k + 1)$: Sortie du processus prédite du retard pur système

Λ : réglage de la forme de la réponse

$u(t)$: entrée du processus

$y_r(t)$: consigne d'asservissement.

Development, testing and evaluation of a night vision goggle compatible BO-105 for night low level operation.

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ABSTRACT

By request of the Royal Netherlands Army Staff, a test and evaluation program was carried out by the Royal Netherlands Airforce. The overall aim of the program was to select and integrate a package of night vision and supporting equipment for the BO-105 C helicopter that will allow round the clock operations in support of the RNL Army, specifically at night at low altitudes. An ex civil BO-105 DB helicopter was used as a testbed in the program. In addition to the original dual pilot IFR equipment (VOR, VOR/ILS, Radar Altimeter and 2 axis Stability Augmentation System), a Doppler Navigation System with Mapreader, a TACAN and recording equipment were installed in the test helicopter. In a pre-evaluation program, two types of helmet mounted Night Vision Goggles (NVG's) were selected for further evaluation. After the 2 axis SAS had been replaced by a 3 axis CSAS and NVG compatible cockpit lighting had been installed in the test helicopter, night low level operational flight trials were carried out. This paper describes the selection of the NVG's, the NVG compatible lighting and presents the pilot experiences and opinions concerning the low level night flight trials. The trials indicated the feasibility of the concept. A selected equipment package will be retrofitted into the BO-105 fleet, with the aid of the airframe manufacturer, MBB. A prototype has been constructed at our Depot at Gilze Rijen Airbase. Flight tests have recently been completed.

1. INTRODUCTION

Up to this day the BO-105 C is being used as an all purpose Light Observation Helicopter for day missions under Visual Meteorological Conditions (VMC) only. Night flight is conducted under Special VFR, for training purposes only. Minimum meteorological conditions for local night flying are 3 kms visibility, 1000 ft cloudbase and 5 kms visibility and 1500 ft cloudbase for cross country flights outside controlled airspace, minimum height above ground is 600 ft.

In 1975 in connection with a night-landing equipment evaluation program in an Alouette III, limited experience was gathered with a pair of 2nd generation AN/PVS-5 NVG's. Although the NVG's showed great potential as a pilots nightvision aid, it was immediately recognized and later confirmed by literature, that, as a result of the limited field of view and resolution of the NVG's , workload and disorientation would be a limiting factor and could become a problem. For this reason it was our opinion that rather than starting with the basic aircraft and adding equipment as required, the test helicopter should be equipped with a low level navigation system with automatic map display and a radar altimeter. All flight and performance instrument dials must be easily readable while wearing NVG's. Also consideration was given to the fact that, if one wants to investigate the limits off the NVG's, inadvertent Instrument Meteorological Conditions (IMC) are likely to be encountered. For this reason the test helicopter had to be certified according to Instrument Flight Rules (IFR).

In December of 1982 a civil, dual pilot IFR, BO-105 DB helicopter was purchased by the RNL Army, this helicopter was to function as a testbed in an equipment evaluation program. First the original civil avionics equipment was qualitatively evaluated for possible military application, later, at various stages in the program certain systems were either added or replaced. To satisfy procurement formalities all systems and equipment, used in the program, underwent comparative testing. To limit the scope of the equipment selection program where possible, first consideration was given to avionics equipment that either was already available in the RNL Airforce inventory, or available as Optional Equipment for BO-105 helicopter (known technology). Most of the avionics equipment under test was installed on a pallet in the baggage compartment of the test helicopter. Equipment installation was done by technicians of our overhaul facility (DVM) at Gilze Rijen Airbase, with assistance of the airframe manufacturer. Concerning the mission supporting equipment, only those aspects that are relevant to the night low level mission will be discussed in this paper.

2. TRIAL OBJECTIVES

The main objective of the trials was, to select and integrate a cost effective package of Night Vision and mission supporting equipment, that will provide the BO-105 C with an around the clock operating capability, in its future role of rearward observation helicopter for the RNL Army.

3. PRELIMINARY EVALUATION

In May of 1984 a preliminary NVG evaluation program was carried out. To be complete and to satisfy procurement regulations, modified drivers goggles were also tested. 5 sets of NVG's were available for testing; Cyclops, (a bi-ocular monotube goggle), AN/PVS-5, MFP (Modified Face-Plate), AN/AVS-6 Aviator Night Vision System (ANVIS), BM 8043, (German Army development) and Cats Eyes, (UK prototype designed for fixed wing aircraft).

The preliminary evaluation was carried out in a fleet BC-105 C, only the cockpit lighting had been adapted in a provisional manner, i.e. some light sources had blue filters taped on, other, less essential, were taped over. The instrument-panel was illuminated by an Electro Luminescent wrist lamp taped under the glareshield. Selection criteria were, weight, balance, stability, eye-relief, comfort, adjustment, alignment and ease of installation and removal.

3.1 The Cyclops mono-goggle.

The Cyclops mono-goggle was originally designed as a driver's goggle, helmet mounting and counterbalance weight were improvised. The bi-ocular goggle consisted of a metal housing, which incorporated a single 2nd gen. Image Intensifying Tube (IIT), a battery case and glass optics. The single image was divided by prisms and presented to the user through two small diameter eyepieces. The Field of View was 40°. Advantages and disad- vantages are listed below:

a. Advantages; Low cost, lightweight, easy to attach and remove.

b. Disadvantages; No alternate power source, difficult battery switching, insufficient eye-relief, no vertical adjustment, no stereopsis, eye discomfort, goggle misalignment, counterbalance weight required.

The disadvantages outweighed the advantages considerably, the Mono-goggle was therefore considered not acceptable for further use in the evaluation program. 3.2 AN/PVS-5 MFP (Modified Face-Plate).

These goggles, were basically a modified version of the well known and widely used driver's goggles. 2nd gen + IIT's were fitted. To aid peripheral vision and allow direct instrument monitoring, the faceplate had been altered (cut out). The helmet mounting attachment were improvised i.e. straps attached onto Velcro pads on top of the helmet. A dual battery pack, originally developed for AN/AVS-6 could be attached to a Velcro pad on the back of the helmet and functioned as a counterbalance weight.

The only advantage, low cost, does not outweigh the shortcomings, i.e.:

a. Instable helmet attachment, causing misalignment of the small diameter eyepieces and subsequent eye discomfort and fatigue.

b. Insufficient eye relief to allow easy instrument monitoring.

c. Excessive weight, an extra balance weight was required to prevent helmet rotation.

The improvements, as compared to the original AN/PVS-5 used in our earlier program in 1975, were considerable, but only acceptable as an interim solution.

3.3 Cats Eyes.

The Cats Eyes NVG's, as shown in Fig.1, were a prototype helmet mounted design, under development for fixed wing aircraft, for use in combination with a Forward Looking Infrared (FLIR) image, displayed in a Head Up Display (HUD). The goggles were attached to an SPH-4 helmet by a quick lock and release unit. The binocular goggles consisted of an all metal housing, which incorporated 2nd gen IIT's, a battery case, and glass optics. The forward end of the goggle was similar to other goggles, the rear end however, was quite different. The image is deflected 90° through a pair of prisms into a pair of small clear glass combiner blocks. The user sees the projected image at infinity, superimposed onto the outside world. The rationale behind this solution is that, through proper use of filtering and electronic switching, the pilot can observe the HUD scene directly through the eyepiece, instead of having to look underneath. The field of view per tube was 30°. To increase the lateral FOV to 40°,- the tubes were installed with a 10° divergence. Red (minus blue) filters were available for installation over the objective lenses for use in combination with blue/green cockpit lighting.

Cats Eyes.
Figure 1.

The most important advantages and disadvantages of these prototype goggles are listed below:

a. Advantages; Good eye relief, excellent peripheral vision and a clear stable image. Corrective spectacles can be worn.

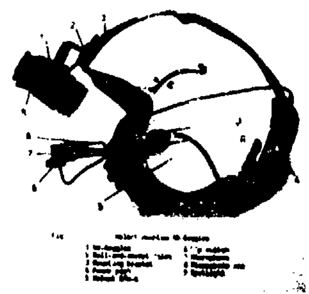
b. Disadvantages; Mounting was difficult, a considerable amount of counterbalance weight was required, double image of bright light sources at close range (projected image did not exactly overlay the real world scene), eye discomfort because of the 10" divergence of the tubes (after a certain period of time, at increasingly short intervals, the pilots had problems to make left and right images overlap).

Because the goggles in question were a first prototype, much can be and has been corrected. Recently, the RNL Airforce had the opportunity to evaluate a more advanced prototype of reduced size and weight, in combination with a FLIR, in a fixed wing aircraft. In this program the feasibility of the concept was demonstrated . This model, however, was not available for helicopter evaluation in the required time frame. Further investigation would be required to determine if having a "See Thru" capability in a helicopter without a HUD, outweighs the extra weight penalty and output reduction of the prism construction.

3.4 AN/AVS-6 and BM 8043.

Both designs were quite similar, as is shown in Figs 2 and 3, both have twin intensifier units attached to the visor cover of the helmet and both use the battery case as a counterbalance weight.

AN/AVS-6
Figure 2.



BM 8043
Figure 3.

Some particulars of AN/AVS-6 were, 2nd gen + IIT's were installed. A special visor construction was required. In flight mounting and removal of the goggle was possible. The visor could be used in flight with the goggles in the stowed position. Red ("minus blue") filters were provided. The housing was made of plastic material. The objective lenses could be focused from infinity down to 25 cm. The eye piece lenses could be adjusted individually over a range from -6 to +2 diopters. The demonstrator goggles showed some signs of wear in the adjustment gears and threads, connections and adjustments were either to tight or to loose, on several occasions a diopter adjustment got stuck.

The BM 8043 system could be clamped onto a standard SPH-4 helmet. The goggles clamped onto the standard visor cover, a battery pack, connected by an armored cord, hooked up to the back of the helmet. The goggles were connected with a ball-and-socket joint onto the clamp on the visor cover. Vertical, longitudinal and tilt adjustments could be accomplished simultaneously in a single joint with clamping screw. The interpupillary distance could be adjusted by pushing or pulling the ocular ends of the goggles with both hands. A small, lipswitch operated, spotlight, mounted in between the tubes, could be used for instrument panel illumination. The focus of the objective lenses was fixed at infinity. The eyepiece lenses could be adjusted individually, over a range of +2 diopters. FOV was 42" and 2nd gen plus tubes were installed.

The results of the pre-evaluation were not conclusive. Only the ANVIS and BM 8043 NVG's had acceptable weight and balance and adequate eye relief. Alignment was not critical because of wide (25 mm in diameter) eye pieces. Corrective spectacles could be worn. Both types were recommended for further testing. The results of this evaluation are presented elsewhere in this paper.

4. TESTBED PROGRAM

4.1 General

The assessment of the complete equipment package was conducted in a BO-105 DB ex civil helicopter, by testpilots, assisted by experienced helicopter pilots.

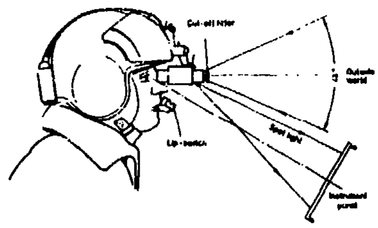
4 sets of NVG's were available for further evaluation, 1 set AN/AVS-6 with 3rd gen. IIT's and 3 sets of BM 8043, 2 sets with 2nd gen + and a set with 3rd gen IIT's. "Minus Blue" filters were available for the latter, the objective lenses of the former were treated with a "Minus Blue" coating. The cockpit of the test helicopter was equipped with several types of NVG compatible lighting. An AN/ASN-128 Light Doppler Navigation System (LDNS) with automatic Map Reader, (K10-0), a 3 axis Control and Stability Augmentation System (CSAS), IFR instrumentation and a Tactical Air Navigation System (TACAN) were also installed.

The program was conducted according the following pattern. First experience and confidence were build up at safe heights in the local area of Deelen Airbase, then, in the well known local low flying area, height clearances were slowly reduced. Next selected low level routes were flown several times, to gain confidence and experience with the use of the Map Reader. Prior to each flight the route was reconnoitered by day for possible newly erected obstacles. Lastly, fairly difficult routes were selected in a relatively unfamiliar low flying area. To insure the safety of flight, prior to each flight, the general area and the boundaries were surveyed in daytime . The meteorological conditions varied from clear sky, full moon, cloud ceiling overcast 200ft, 1500 m visibility in rain and snow showers.

4.2 Cockpit Lighting Evaluation.

4.2.1 Dark Cockpit.

The lipswitch operated spotlight mounted in between the tubes of the BM 8043 goggles Figs 3 and 4, was a German Army Aviation requirement, their mission requires a "Dark Cockpit".



BM 8043 with Spotlight
Figure 4.

During the initial phase, we attempted to operate in a completely blacked out cockpit, none of the pilots felt comfortable, mainly because all flight- and performance information a pilot unconsciouly uses for the execution of the flightwas not readily available. A cross-check of the instrument panel always required his full attention. With the experience level of the pilots in mind and because the BO-105 task does not require covert missions, further investigation into the "Dark Cockpit" concept was abandoned.

4.2.2 Floodlighting.

Several types of floodlights were available for evaluation, (1) Micro-louvered E.L. lamps, mounted under the glare shield, (2) a blue Kopp #0005 filter over the Utility light in the overhead console and (3) an Ultra Violet (U.V.) lamp, mounted onto the overhead console.

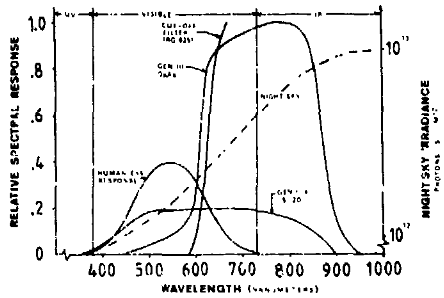
Options 1 and 2, required relatively high light levels, this caused high lights on the instrument panel and reflections in the canopy. Large shadows were cast over the face of the instrument panel leaving important sections barely readable. Option 3 appeared to be the best solution of the 3. The only problems were that light source had to be moved quite close to the instruments and that only those symbols, that had been treated with fluorescent paint, were visible. The effect however looked good, even colours showed up.

Because U.V. light is invisible to the human eye, U.V. floodlighting is an ideal solution, if covert missions are required. Another advantage of U.V. light is that the frequency lies far outside the sensitive range of NVG's, "Cut Off" filters are not required. A disadvantage is that all instruments have to be treated with special paint, this may become cost prohibitive.

4.2.3 Blue Cockpit lighting.

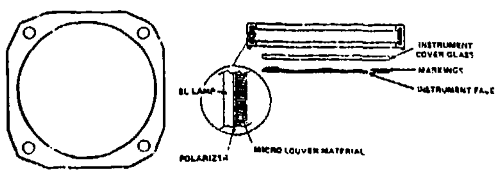
A combination of blue Electro Luminescent (E.L.) lamps and blue Kopp #0005 glass filters was used to make the cockpit NVG compatible. To avoid goggle "Shut Down", red ("Minus Blue") filtering of the NVG's was required. Glass filters (RG 645 and RG 665) were available for the BM 8043 goggles, the objective lenses of the 3rd gen. AN/AVS-6

goggles had been treated with a "Minus Blue" coating. Both options functioned well, reflections in the cockpit windows were suppressed. We were however surprised to find out that 3rd gen goggles also required a "Cut Off" filter. Figure 5 shows the relative sensitivity curves.

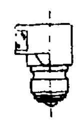


Relative sensitivity curves.
Figure 5.

Most 3" and 4" dials in the instrument panel were fitted with E.L. Bezel lamps (Fig.6), for the rest of the 3" and most of the 2" dials, blue filtered Post lights (Fig.7) were used. The readability of the 3" and 4", E.L. Bezel equipped, instruments was excellent, the dials stood out clearly against a dark background. The light was spread evenly over the whole face of the dial, coloured limit markings were highlighted. E.L. Bezels were not suited for 2" instruments and 3" instruments with deep lying dials, because symbols close to the rim of the dial were obscured. Blue filtered post lights gave satisfactory results when used in combination with most 2" instruments, the position and the direction of the light beam however, have to be selected carefully.



E.L. Bezel
Figure 6.



BOLT LIGHT CONVERSION CAP
Figure 7.

The text windows of the Caution Panel lights were replaced by blue Kopp filters with the text silk screen painted onto it in negative black letters. Day and night readability is good, some doubt exists however if pilots, who are conditioned to react on red and amber warnings and cautions, will recognize an emergency promptly. A cancellable red Master Warning light will be installed in the Fleet helicopters.

A special problem was formed by the Steering Hover Indicator Unit (SHIU) of the LDNS, it had several different types of incandescent lights that had to be made NVG compatible, 2 amber caution lights, a white indicator light, white internal lighting and amber an digital display. After several unsuccessful attempts with many yards of tape, a filter cap of blue Kopp #0005 filter material was made up, that could be fitted over the whole instrument. As an interim solution it functioned well. Up till now no other acceptable solution has been found.

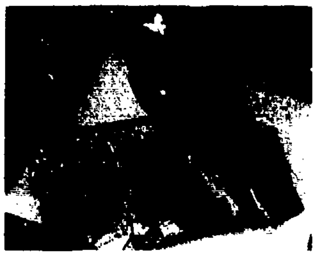
All panel lights in the overhead- and center console were disconnected. The display windows of the LDNS, Control and Display Unit (CDU) and the Radio control panels were replaced by blue Kopp #0005 glass windows. The readability of the filament displays at night was excellent, in bright sunlight it was acceptable. The modification will be applied to the fleet. Blue filters will be used in some of the existing incandescent control panels (IFF, ICS).

Presently the implementation of blue, green and yellow Light Emitting Diodes (LED's) is under investigation. As can be seen from the curves in Figs 5, the use of a red "Cut-off" filter at about 630 Nanometers allows the use of E.L. lamps as well as LED's. The feasibility has been demonstrated laboratory tests. LED's are inexpensive and available in a large variety of shapes and sizes. Many applications appear to be possible e.g. Alpha numeric displays, Caution panels, keyboards etc.

A combination of E.L. Bezels and filtered Post-lights has been recommended, cockpit layout and illumination is being optimized for the mission during prototyping. LED bezels and LED overhead-panel lighting are presently under development.

4.3 Automatic Map Display

The Automatic Map Display (AMD K10-0) (Fig.8) had no integral lighting, a hand held lamp had to be used. A provisional modification proved to be quite successful. A large sheet of E.L. material, the size of the display window, was taped onto the top of the Map Display. A map placed over the E.L. lamp becomes a NVG compatible transparency with excellent readability, only a minute colour shift is evident. Terrain features such as contourlines, roads and railways show up clearly. With the lid closed in the normal fashion, the cross hairs present a clear position indication without shadows. A reostat was installed for dim control.



K10-0 Map Display.
Figure 8.

K10-2 Map Display
Figure 9.

Modified in this fashion, the Map Display became a very useful tool. With the Map Display integrally illuminated, the co-pilot/navigator had an extra hand free for writing and map changing.

By the end of 1985, the manufacturer informed us that the K10-0 would not go into production and offered us an improved version, the K10-2. The differences were significant. Instead of cross hairs, a moving light dot of variable intensity was used for position indication. Improved electronics, map preparation and map changing made the use of prepared maps easier. Integral NVG compatible lighting provided excellent readability. Although the K10-2 Map Display is still rather bulky it was recommened because to our opinion it is considered an essential tool for safe low level NVG operations.

4.4 External lighting.

The landing lamp was fitted with an IR filter, the light intensity was too high for NVG operations. It was found the a white navigation light of variable intensity provided sufficient light for NVG landings and maneuvering in unprepared landing zones.

Navigation and formation lights have to be defined at a later stage of the program, in Field Trials, after a sufficient number of modified helicopters has been introduced in the fleet.

4.5 Final Goggle evaluation.

The BM 8043 and the AN/AVS-6 functioned well in the blue cockpit, a day VMC type cockpit cross-check could be adopted. Until late in the program the pilots had no particular preference for either type. Some found the BM 8043 easier to focus. All pilots liked the break away coupling of the AN/AVS-6. First of all, from the flight safety point of view and second, because it allows the user to stow or remove the goggles in flight if so desired. Another useful feature was that the Visor was operable, with the goggles in the stowed position. This feature is particularly useful when flights are performed around sunrise or sunset, with the sun just over the horizon. Towards the end of the flight trials another model of the BM 8043 with "Break Away" coupling became available for testing. A few minor modifications to the goggle- and visor assembly made the use of the Visor possible. Because the design of the BM 8043 is sturdier than that of the AN/AVS-6, ANVIS goggles and less sensitive to temperature changes and because the Lythium battery pack offers longer operating time, with triple redundancy, this modified version was recommended. Figure 10 shows the modified BM 8043 NVG's.

BM 8043 Modified.
Figure 10.

4.6 Goggle performance.

The performance of the 2nd and 3rd generation Image Intensifying Tubes (IIT) was qualitatively evaluated. At the start of the trials existing light levels were measured on the ground. We soon abandoned the procedure because the measured values at ground level were not representative for those that were encountered in flight. All pilots preferred the 3rd generation IIT's, they had better resolution, less noise and better performance under marginal condition. One set of 3rd gen. IIT's was of particularly poor quality, looking through these goggles was like looking through a dirty window. In spite of this, under marginal light conditions they performed better than the best of the 2nd gen. + IIT's. 3rd generotion IIT's were recommended.

4.7 Stability Augmentation.

The 2 axis SAS, the testbed helicopter originally was equipped with, only provided short term rate damping and could only be used in level cruise flight. We required a "Fly Thru" system that was optimized for low speed, low altitude maneuvering. The only other BO-105 certified system that was available at that time was a 3 axis, attitude referenced Control and Stability Augmentation System (CSAS). The system was duplex in the roll and pitch channels. It could be operated in 2 modes, the Attitude Hold (ATT) mode, it provides long-term attitude retention and the Stability Augmentation (SAS) mode, it is a "Fly Thru" mode that provides short term attitude damping during manual flight. A rate damper and a "Force Gradient" with Mug Brake were used in the yaw axes. The Yaw SAS was designed for feet-on-the-floor operations in cruise flight, the system had to be switched off before landing. To allow the use of the Yaw SAS during low speed maneuvering flight, the gradient of the pedal forces was altered.

The system eliminated the dynamic instability in pitch and reduced most of the control cross coupling effects that are inherent to "Hingeless" rotors. Specifically the low airspeed flying qualities were greatly improved, the reduction in pilot workload, when flying with NVG's at low altitude and in turbulence was considerable. The ATT mode was not intended to be used as a "Fly Thru" mode. During the low level navigation flights, we tried it anyway and found that the increased positive centering of the cyclic stick and the automatic return to wings level attitude reduced workload on long stretches even further. The improved stability and qualities are also satisfactory for IFR flight. Because the BO-105 mission mainly involves dual crew operations, a single channel 3 axis SCAS was recommended.

4.8 Low Level Navigation.

For low level navigation an AN/ANS-128 LDNS was selected for evaluation. Up to 10 Waypoints could be entered into the Control an Display Unit (CDU). The CDU displays navigation data in a digital form. The CDU drives the Steering Hover Indicator (SHIU), in the NAV mode, it displays distance, left-right steering information and ground speed. This information is particularly useful for point to point navigation, it enables the crew to roughly estimate the position in relation to the destination. For contour flight navigation at night, accurate position information is essential. Without any additional equipment the co-pilot/navigator still had to keep track of his position on a hand held map. Only flights over memorized routes were considered safe.

The key to routine night low level operations was the Illuminated Automatic Map Display. It allowed the co-pilot/navigator, to monitor his position instead of having to plot it continuously. Fast position updates on the map on any outstanding terrain feature were possible. We only had 1:50.000 and 1:250.000 scale maps of the area available. The latter did not have enough detail and the former had to be changed quite frequently. We hope that 1:100.000 scale maps with just enough detail to avoid clutter, will soon be available. Another improvement that would increase the value of the Map Display would be an interface that allows automatic update of the LDNS by means of a position update on the Map Display. Another welcome improvement would be a drastic reduction in weight and volume.

5. Summary.

With the recommended equipment package it was demonstrated that a BO-105 helicopter could be operated by experienced helicopter pilots low level at night over fairly unfamiliar terrain under adverse weather conditions.

The helmet mounted NVG's gave us a night low level capability, the 3rd gen IIT's added an extra darkness and reduced visibility margin.

The blue NVG cockpit lighting made easy monitoring of flight and engine performance instruments possible and allowed the use of navigation equipment. This greatly improved confidence and reduced workload.

The Illuminated Map Display improved navigation accuracy, reduced the workload of the navigator, and allowed him more "Eyes Out the Cockpit" time, with added flight safety.

The CSAS improved the overall flying qualities in cruise, in low speed maneuvering flight and in turbulence. Pilot workload was greatly reduced.

The IFR instruments in the test bed helicopter made IFR recovery possible in those instances that the weather deteriorated below NVG capabilities. The addition of a TACAN was recommended, to provide the BO-105 with a military IFR capability, this will add to the safety of flight during NVG operations under marginal weather conditions and greatly increase its "Round The Clock" operating capability.

6. BO-105 C Mid Life Update Program.

Update Program has received Parliamentary approval. In February 1986 the construction of a prototype was started at our overhaul facility (DVM) at Gilze Rijen Airbase, with assistance of the airframe manufacturer MBB. The flight testprogram has recently been completed. With the assistance of system engineers of MBB and Sperry the systems were tested and tuned in flight. With the exception of the cockpit lighting, all systems functioned according to specification. The cockpit lighting evaluation could not be carried out due to late delivery by the manufacturers. By the end of 1986 the retrofit program is planned to start. In this program the following modifications will be carried out:

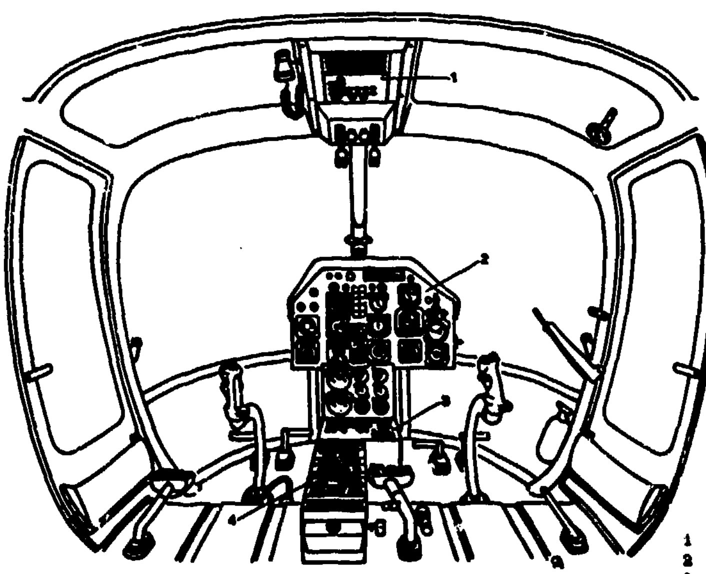
a. Redesign of the pedestal shape, to increase forward field of view (Fig.11). Complete redesign of the cockpit layout and electrical system. This includes co-pilot IFR instruments and NVG compatible "Blue" lighting.

b. Installation of S.E.L. AN/ASN 128 Light Doppler Navigation System, G.E.C. Avionics TACAN, King KN 405 Radar Altimeter, Sperry Helipilot, single channel, 3 axis CSAS and NVG compatible external lighting.

Additionally the procurement of a number of BM 8043 Mod. with gen 3 IIT's and K10-2 map Displays have been ordered.

7. Recognitions.

We like to express our appreciation to all those manufacturers, that allowed us to evaluate their equipment, for their welcome advise and support. We also like to thank the Royal Aircraft Establishment, Farnborough and the ATV, Buckeburg, from whom we obtained a considerable amount of valuable advise.



Redesigned Pedestal
Figure 11.

A129 ADVANCED SOLUTIONS FOR MEETING TODAY'S COMBAT HELICOPTER REQUIREMENT

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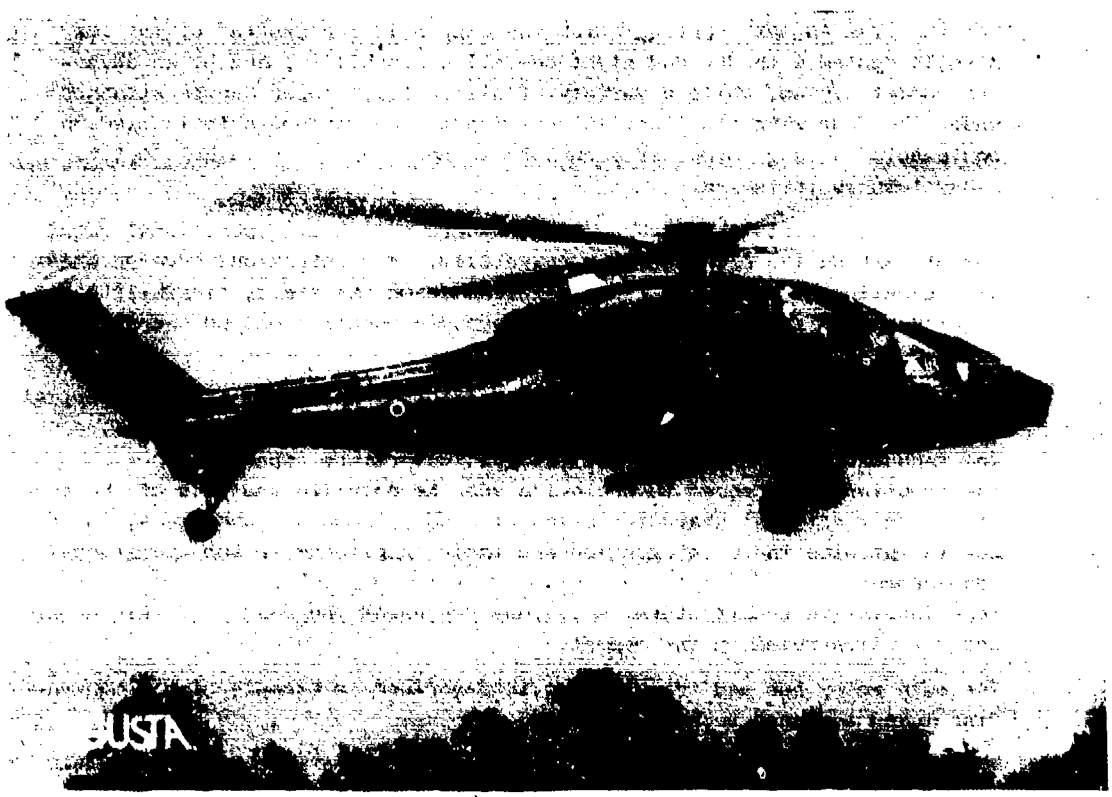
SUMMARY

- The A129 is a light attack helicopter developped for the ITALIAN ARMY in the anti tank configuration.
  The A129 has a four blades articulated main rotor compatible with Mast Mounted Sight (MMS) installation; it is a twin engine helicopter with tandem cockpit configuration provided with armored and crashworthy seats and an environmental control system integrating an air filter.
  The fuselage and related installations, as the landing gear and the fuel tanks are designed to withstand loads arising in case of crash.
  For the ITALIAN ARMY configuration the weapon is represented by the TOW missile system with day and night operative capability, and by an advanced rocket system, while a variety of alternative weapon can be attached under the four wing stations. In addition to the various avionic mission equipments, a large computer structure integrating the main helicopter subsystems is installed.
- The A129 represents the answer to the modern and severe operational requirement, asking the A/C maximum survivability, day, night and adverse wheater operational capability, easy maintenance on the field, flexibility for a quick weapon reconfiguration and affordable acquisition and operative costs.
- The key factor to achieve all these requirements has been the technological innovation considered from the early phase of the design.
  The technological innovation is based on two primary aspects:
  the adoption of advanced technologies and the detailed analysis of the physical and functional characteristics of each helicopter subsystems, in order to optimize their integration and their compliance to the operational requirement.
  Some innovative architectural solutions represent the result of this work, and are illustrated in the report.
- The main rotor hub end transmission is described as example of not conventional and advanced technology design.
- The engine installation represents the case of a configuration where the operative requirement induced a deep physical integration among the subsystems.
- The large computer structure, or INTEGRATED MULTIPLEXING SYSTEM (IMS) installed on the A129, based on the multiplexing and microprocessors technology, constitutes a major example of functional integration between the helicopter and mission subsystem.

INTRODUCTION

The Agusta A129 is a light combat helicopter, developped under commitment of the ITALIAN MINISTRY OF DEFENSE.
The first production batches will be operated by ITALIAN ARMY primarily in the antitank configuration.
The A129 has a single four blades articulated main rotor compatible with a Mast Mounted Sight (MMS) installation, and a two blades semirigid tail rotor.
It is powered by two Rolls Royce MK1004 engines of about 1000 Shaft Horse Power (SHP) each, located in an engine installation designed for operations in severe ambient conditions including a particle separator and a low Infra Red emission exhaust system.
The drive system consists of a main transmission, with high speed engine input shafts, and a tail transmission, with supercritical shafts and grease lubricated gearboxes.



A129 HELICOPTER IN ANTITANK CONFIGURATION

The fuselage has the forward section dedicated to the cockpit area, with a tandem crew configuration where gunner-copilot seat is located in the forward position, and pilot in the rearward one.
On the nose is located the sight of the antitank TOW missile system, with day and night operative capability, above which is set the infra red sensor of the pilot night vision system.
Crew is protected by armored and crashworthy seats.
An environmental control system is included in the basic A/C configuration and can be integrated with a biological and chemical filter to protect crew from the external contaminated air.
In the central part of the fuselage are installed two crashworthy and self-sealing fuel tanks.
The landing gear is derigned to withstand limit, hard landing and crash loads, absorbing the most kinetic energy during an impact of the aircraft with the ground.
About the weapons and mission equipments, the A129 in the anti tank configuration for the ITALIAN ARMY will have the TOW missile system as primary weapon, and an advanced rockets system as alternative or complementary one.
The avionic package will include communication and navigation systems, aircraft survivability equipments as radar and laser warning, and radar and infra red jammer, visionic system for night operations, based on Flir sensors and Helmet Sights and a large computer avionic structure that menages and integrates the mission equipments and the aircraft subsystems.

A129 and the modern combat helicopter requirement

The A129 represents the result of an iterative joint work started in 1972 between ITALIAN MINISTRY OF DEFENSE and AGUSTA, to define modern operational requirement for combat helicopter from the military side and a technical solution from the industrial side.
Sintetically speaking, the main and driving operational requirements are:
- Suitability of the weapons and mission equipments for the required role(s)
- Survivability of the aircraft during the operations, in a scenario where a sophisticated threat is present, as IR and Radar guided missiles, biological and chemical contaminated atmosphere, small caliber weapons, etc. Survivability induces the A/C to have a noticeable agility, to present a low detectability or low visual acoustic radar and IR signature, plus the installation of electronic warfare warning and jammers, to show a good ballistic tolerance and installations against contaminated atmosphere, and to provide adequate crashworthy safety margins.
- Capability of operating during adverse weather and ambient conditions, day and night

Other important requirements are:
- The capability of operating from the field (first level) with minimum support showing an high degree of availability
- The total acquisition and operative cost of the whole helicopter-weapon system: or, in other terms, its affordability.

In addition it must be outlined the requirement of flexibility of the helicopter for a weapon and mission equipments reconfiguration, in order to improve the operational flexibility using the same aircraft for other combat roles as scout, escort etc.
The Mast Mounted Sight (MMS) and the turrett gun under the nose provisions are good examples of installations already considered during the development phase of the A129.

Strictly tied to the above requirement there is to be mentionned an emerging trend that requires the development of the helicopter family; that means helicopters of similar size, able to satisfy different roles and generated from the same design as variants or derivatives, with the maximum commonality among them.
Example is the developement of the battlefield support helicopter derived from the combat A129 maintaing the same dynamic components.

Technological innovation applied to the A129

The technological innovation has been considered during the design phase as a key factor to reach the technical goals set by the operational requirement.
The technological innovation on A129 is based on two main factors:
- the adoption of advanced technologies
- the detailed analysis of physical and functional characteristics of each subsystem in order to optmimize their integration adressed to the operational requirement compliance.

  The marriage between these two aspects originated some innovative architectural solutions representing optimum/trade-off answers to the requirements.

Some aircraft subsystems are here illustrated as examples of such technological innovation.

1. Main Rotor Pylon (Rotor and Transmission)
   . The driving requirements for the main rotor of the A129 are:
     Agility and handling qualities
     (adequate control power at high and low G)
     Low vibratory levels
     Survivability: detectability and vulnerability
     Operation in severe external environment (wind and icing conditions)
     MMS compatibility
     High degree of RAM (Reliability, Availability, Maintainability)
   . The main rotor is fully articulated, with four blades in composite material attached to the hub assy through composite grips.
     In each grip is located a single elastomeric bearing that allows the blade rotations along the three flapping, lead-lag and pitch axis. (See fig. 1.1)
   . Three main design goals have been achieved by this main rotor lay-out, that shows a noticeable improvement with respect to the conventional rolling bearing rotors:

     Articulated rotor =
     The rotor is kept as pure articulated in flight, around a single virtual hinge because this architecture provides low vibratory levels transferred by the blades to the A/C while gives a suitable rotor control power for the A/C agility, with a proper value of the hinge offset.

     Simplicity =
     The design of a single elastomeric bearing for each hub arm induces a great simplicity, with a large reduction of the number of the hub components, and relevant benefits of weight and manufacturing cost.
     Furthermore, the "On Condition" criteria for maintenance is applicable for the most rotor components, cutting down the operating cost and improving the maintanability and availability characteristics.

     Safety =
     In general, a damage to a metallic rotor component can seriously jeopardize the flight safety, for the high speed of the crack propagation.

The damage can be originated by a defect of the component (quality or servicing problem) or by an external reason as a ballistic threat.
The use of composite and elastomeric materials allows the design of components with very low crack propagation speed and many safe flights are possible after the appearance of the damage; a great level of safety is thus achieved.

. The main rotor blade, manufactured in composite material, has been designed for optimizing aerodynamic performance, low acoustic noise signature, and ballistic tolerance till to 23 mm hit.

. The rotating controls of the main rotor have been designed to be compatible with a Mast Mounted Sight installation. (See fig. 1.2)
The control rods connect the swashplate assy located above the hub plane to the servo actuators set at the center of the transmission assy.
The four reduction stages transmission is of modular design and it is attached to the a/c fuselage by means of a redundant pylon support system.

. The main transmission presents a peculiar annular geometry, having inside the support to the servo actuators.
The design of this four reduction stages transmission is oriented towards the modularization: it is structured in several subassemblies, each of them is monitored by a dedicated systems.
For logistics and ballistic protection many subassemblies are directly installed on the transmission assy:
- the two hydraulic power supply groups, physically seperated each other
- the transmission oil cooling system
- the electric alternator
- the compressor for the crew environmental control system

that allows a minimum number and lenght of hydraulic lines.
To further improve the accessibility to the pylon, the air intake structure is hinged, and can be opened as a large door.

2. Engine installation of the A129 (See fig. 2.1)

The engine installation of the A129 is a good example of a system integration, developped to satisfy the following operational requirements:
- minimum vulnerability, that means:
  low detectability, that is achieved by minimizing the IR signature from the engine bay and exhaust gasses
  low vulnerability to ballistic damage, that is achieved:
  by the choice of a twin engine configuration,
  by the definition of an emergency power level applicable to the engine and transmission in case of the other engine failure
  by a proper positionning of the two engines, set at a suitable distance each other and ballistically separated
  by a fire resistant engine bay
  by a redundant engine support system, able to withstand the loads arising in case of crash
  by the design of a ballistic and crash tolerant dual fuel tanks, connected to the engines by independent fuel lines, with the fuel pressure continously monitored by the on board computer able to manage the fuel valves in case of ballistic damage.
- The operativity in severe ambient conditions, that is achieved:
  by integrating into the air intake an inertial engine particle separator (FOD and sand)

by developping an engine air intake able to operate in icing conditions

- The Supportability on the field, that is achieved:
  by an easy accessibility to the engines for cecks or removal,
  by avoiding any need of alignement procedure or tooling when the engine is installed on the helicopter
  by the use of an integrated engine, where all the engine accessories as the oil cooling system, the Electronic Governor, the suction fuel system are packed into the engine
  by a modular engine architecture, where the most of the engines modules are physically and functionally interchangeable
  by an engine health monitoring system installed into the helicopter, that continously monitors the proper engine behavior in flight, automatically menages the caution and warning messages in case of malfunction or limit exceedance, and record the engine usage for maintenance purpose.
  In addition this system allows crew to perform an instantaneous engine power ceck within the helicopter flight envelope.
- Minimum penalties (weight and power losses) to accomplish all the above functions.

3. Integrated Multiplexing system (IMS) on the A129

. The A129 is provided with an avionic integrated structure based on the microprocessor and multiplexing technology using a data bus according to the MIL 1553B.
This redundant and centralized processing structure enables an efficient exchange of informations between the basic A/C subsystems, mission equipments and crew.

. This technical solution is the answer to the mission requirements, asking the avionic structure:

- to maximize the mission success probability
- to provide adequate safety margin in case of failure or ballistic damage
- to maximize the availability of the A/C, reducing the maintenance time
- to reduce the physical penalties, in terms of room and weight,for the installation of the avionic structure and, finally,
- to reduce the cost, and improve the flexibility for the future integration of other mission equipments.

In general, a conventional avionic architecture requires, from the Hardware aspect, dedicated sensors, dedicated controls and displays and processors.
No redundancy is provided, and the mission success probability is depending from the product of the reliability of each critical function.
In the case of the A129, a redundant computer and data bus is provided, with cross strap autoreconfiguration capability; thus a relevant higher value of mission success probability is achieved.
Another important characteristic of this architecture is that all the data collected from the sensors or processed by the computer are available both to pilot or copilot gunner.

. Design considerations required an information linkage between subsystems accomplished by a sistem very safe and reliable, while the reconfiguration, the performance vs. weight ratio, the noticeable accuracy required by the functions and the flexibility requirement oriented the technical solution towards a digital technology. (See fig. 3.1).
The avionic structure of A129 is based on two equals computers, each of them operating on the same data and provided with autodiagnostic capability.

The computers are connected through a redundant bus 1553B to remote units collecting from the subsystems sensors the data, concentrating them on the bus, and properly distributing the results of the processing steps.

- The interface of the computing structure, called integrated multiplexing system (IMS) with the crew consists of two devices installed on the cockpit:
  - a multifunction display unit which is basically an interactive video display terminal
  - a multifunction keyboard .
- About the physical aspects, the IMS consists of four black boxes (two master units containing the processors and two remote units) two keyboards and two multifunction CRT displays.
- The IMS has been designed to perform the following functions:
  - CNI (Communication Navigation and Identification) control and management
  - AFCS (Automatic Flight Control System), including basic stabilization and higher modes of the stabilization and control augmentation system (SCAS), coupled with the navigation system sensors.
    Flight director functions are possible, for operational use, as:
    - -- Flight plans (definition and AFCS coupling)
    - -- Attack mode, maintaining the attitude altitude and heading for the weapon release
    - -- Course and hover hold
    - -- Vertical and forward ground/airspeed hold
    - -- Integration with the A/C flight controls in order to make possible a FLY BY WIRE mode for main and tail rotors.
  - Weapons control (advanced rockets, turret gun)
  - Aircraft performance monitoring
  - Caution and Warning management
  - Symbology generation for pilot and copilot presented on the helmet sight or displays
  - Operators interface
  - Autodiagnostic and diagnostic of the equipments connected to IMS
  - Engine and fuel system management and monitoring
  - Electrical power distribution management and control
  - Hydraulic and transmission systems management and monitoring
- An important operational result has been achieved by the installation of the IMS on A129, that is the heavy work load required to the crew by the combat mission is alleviated by the on board avionic computer structure. It allows the crew to concentrate on the mission accomplishment rather than the management and monitoring the helicopter and mission subsystems.

FIG. 1.1

A129 MAIN ROTOR HUB

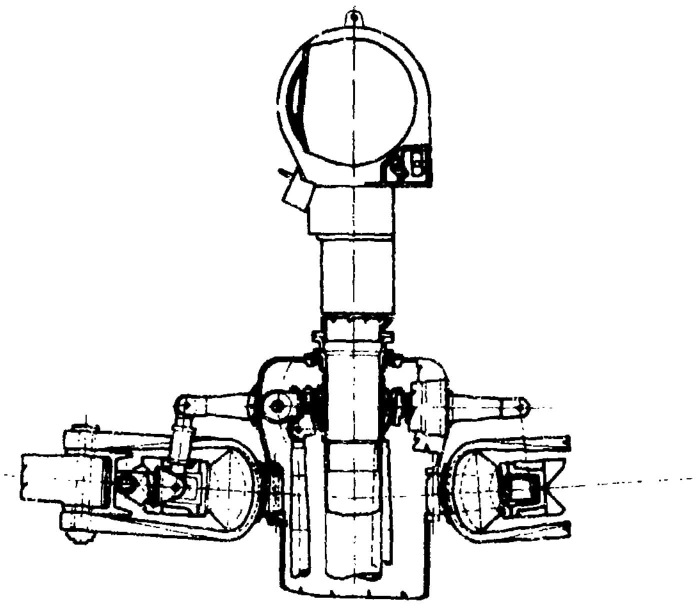
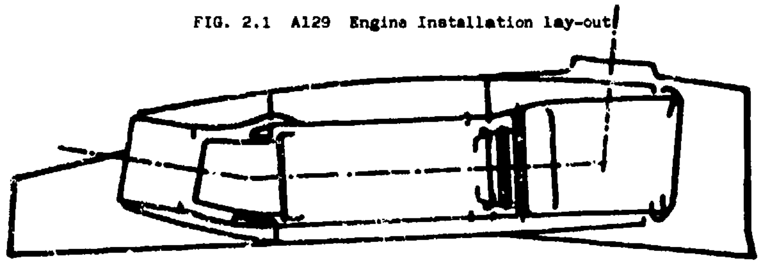


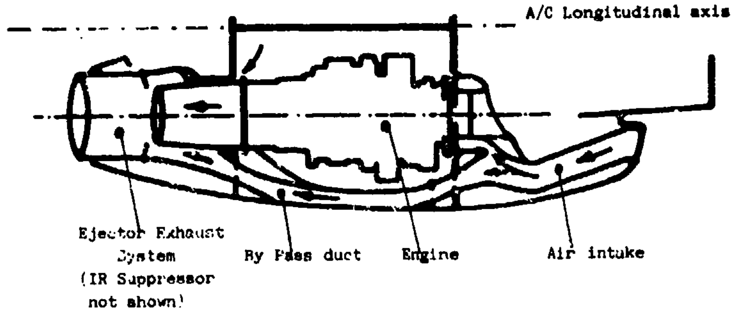
FIG. 1.2

A129 M.R. HUB and Rotating Controls with MMS installation

FIG. 2.1 A129 Engine Installation lay-out



Side view



Plan view

IMS SYSTEM

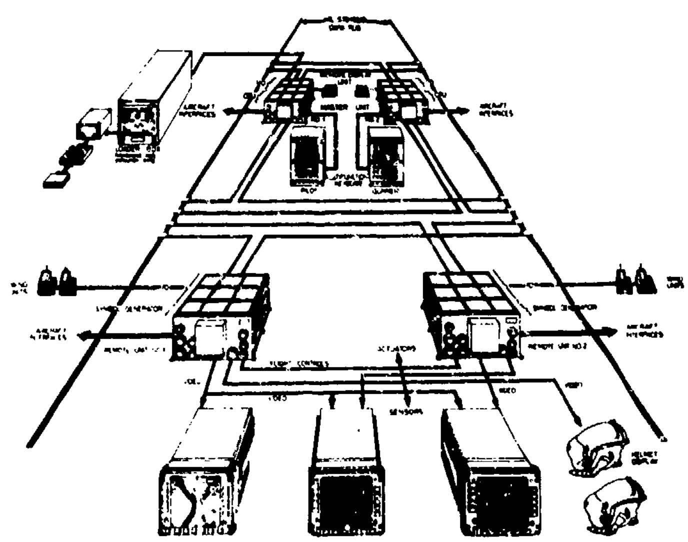


FIG. 3.1 A129 Integrated Multiplexing System (IMS)

APACHE FOR THE BATTLEFIELD OF TODAY AND THE 21ST CENTURY

by

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SUMMARY

The AH-64A Apache is in service with the United States Army. This paper documents the operational requirements that drove the design of the Apache and caused the particular combination of attributes possessed by the aircraft. Attributes to be discussed include performance, survivability, crashworthiness, transportability, deployability, navigation, target acquisition and weapon systems.

The second part of the paper documents the changes emerging in operational requirements, the threat and technology and the probable impact of these changes on the Apache of the future.

INTRODUCTION

On the integrated battlefield of today and the future the AH-64A "Apache" Advanced Attack Helicopter, is a valuable force multiplier and provides an added lethal dimension to the combined arms team. The AH-64A provides an immediate response to the needs of the ground commander. The first attack helicopter ever developed specifically for day, night and adverse weather anti-armor missions, it has the ability to fight and survive at the forward edge of the battle area.

The AH-64A, shown in Figure 1, is a twin-engine four-bladed helicopter operated by a tandem-seated crew of two which delivers unprecedented firepower quickly and with accuracy. The pilot is located in the rear cockpit, placing the copilot/gunner (CPG) in the forward position where he can concentrate on detecting, engaging and destroying enemy targets with an array of weaponry including: HELLFIRE missiles, aerial rockets and the McDonnell Douglas Helicopter Company 30mm Chain Gun® Area Weapon.

FIGURE 1

To enable the Apache to fly and fight at night and during periods of reduced visibility, a unique Pilot Night Vision Sensor (PNVS) and Target Acquisition and Designation Sight (TADS) was developed and integrated to permit navigation and precision attacks under low visibility and night battlefield conditions. The Apache's mission roles include anti-armor, covering force, flank security, economy of force and airmobile escort. The AH-64A provides:

- Superior air vehicle performance
- Day, night and adverse weather operations

- Multiple firepower options
- Twin-engine reliability
- Improved combat survivability
- High reliability, availability and maintainability
- Rapid rearm and refuel capability
- Self-deployment
- Air transportability

In 1981, three YAH-64's, Apache prototypes, participated in a 3-month exercise called OT-II (Operational Test II) in which all weapons and systems were field tested by the U.S. Army under operational conditions. The results of this exercise proved that the YAH-64 was ready for production. In July 1981, Mesa, Arizona was selected as the site for the company's new 240,000 square foot assembly plant for the AH-64. On 15 April 1982, a production contract was signed for the first 11 Apaches, the first of which rolled out on September 30, 1983. The production rate is stabilized at 12 per month. Aircraft number 150 has been delivered to the Army. The Apache has been fielded to combat units giving combat commanders around the clock attack helicopter capability never before achieved.

In the meantime, however, much has changed. The U.S. Army has adopted Army 21 as its new battle doctrine. The threat is increasing considerably in type and level of sophistication and new technologies are developing which give us the tools to adapt to the changing environment. For these reasons, the U.S. Army and McDonnell Douglas Helicopter Company are taking steps to assure the projection of Apache's superior effectiveness far into the future.

The following paragraphs first briefly discuss the operational needs that drove the design of the AH-64A and then discuss how changes in battle doctrine, the threat and technology will result in updates to the Apache to maintain its effectiveness.

U.S. ARMY MISSION

ATTACK

The primary mission of the AH-64A requires engagement and defeat of enemy armor in day, night and adverse weather. The Apache is required to detect, recognize and engage multiple targets, and operate around-the-clock in adverse weather. It denies enemy forces the normal advantage of night, adverse weather conditions and superiority of numbers.

AIRMOBILE ESCORT

These missions include covering force, flank security, economy of force, airmobile escort and area suppression.

The covering force mission is to detect, impede and disrupt enemy forces approaching the main battle area.

The flank security mission is to protect the flanks of the main body forces from surprise attacks.

The economy of force mission is to employ small, mobile forces in strategic areas so ground forces can be massed at the decisive time and place elsewhere on the battlefield.

The airmobile escort mission entails the destruction or suppression of enemy personnel, material and air defense in support of heliborne operations.

AREA SUPPRESSION

In a typical area suppression mission, infantry is transported by tactical transport helicopters to seize or control terrain. The AH-64A's ability suppress enemy air defenses help ensure mission success. An added advantage is the Apache's ability to penetrate enemy defenses along a controlled corridor and neutralize river crossings or other strategic locations ahead of a rapidly moving armored column.

TACTICS

To perform the above missions in a high threat combat environment, Apache utilizes the surrounding terrain to mask itself while preparing to engage and defeat enemy threats. Inherent agility and a large maneuver envelope are required for superior nap-of-the-earth flight, minimizing exposure to enemy weapon systems. Rapid masking and unmasking during attacks required exceptional hover performance and engine power availability.

The AH-64A may engage targets autonomously or work as a team member. As many as 10 target locations can be passed to the AH-64A. The fire control computer can then pre-position weapon systems and display steering information, providing rapid target engagement.

DESIGNED TO THE REQUIREMENT

The following paragraphs highlight the design features of the AH-64A which were driven by the operational requirements described above. A more detailed general description can be found in Reference 1.

The Apache is 48 feet in length and has a primary mission weight of 14,445 pounds. The helicopter is powered by two General Electric T700-GE-701 turbine engines rated at 1690 shaft horsepower each. The

main rotor system is fully articulated with four main rotor blades and four tail rotor blades. At primary mission weight, the Apache exceeds the minimum design requirement of 450 feet per minute vertical rate of climb at 4000 feet, 95 degrees F.

WEAPON VERSATILITY

The weaponry includes laser-guided HELLFIRE missiles, 30mm Chain Gun automatic cannon and 2.75 inch Hydra folding fin aerial rockets. Some examples of the armament options available to satisfy various operational requirements are indicated by Figure 2.

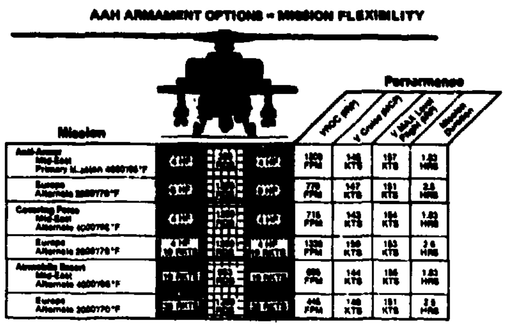
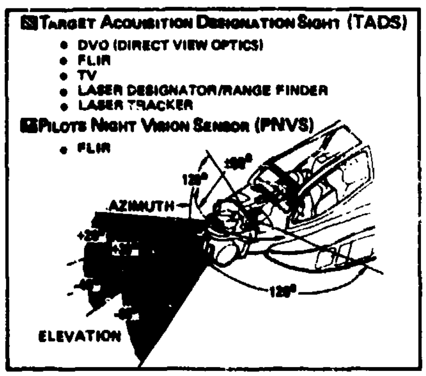


FIGURE 2

NIGHT OPERATIONS

The integrated system that allows the Apache to fly and fight at night and in reduced visibility is the Pilot Night Vision Sensor (PNVS) and the Target Acquisition and Designation Sight (TADS). For the AH-64A, the decision to place the copilot-gunner in the front seat and the pilot in the rear seat was simplified by the desire to incorporate direct view optics into the TADS for enhanced daytime long range target recognition. The optical relay tube transmits the direct view optics to the copilot-gunner station.



TADS and PNVS

FIGURE 3

A laser tracker incorporated in the TADS provides the copilot-gunner with the capability to rapidly acquire targets designated by other ground or airborne designator systems.

An optical relay tube (ORT) provides the TADS controls and display necessary for the copilot-gunner to operate the system. In the head-down mode, the copilot-gunner can view the two field-of-view direct view optics or a cathode ray tube which displays either the two field-of-view day television or three field-of-view FLIR video. Additionally, the head-down mode allows him to view symbology for target engagement and an alphanumeric display which provides fire control system status information. In the heads-out mode, he can view a panel mounted CRT which also displays day television of FLIR video.

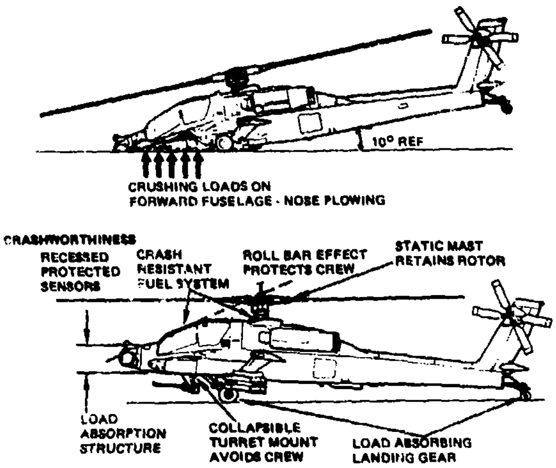
The PNVS allows the pilot to fly the AH-64 in the hostile night environment using terrain flying techniques. A real-time, passive "thermal image" of the "world" outside the cockpit window is displayed on a helmet mounted display (HMD) which the pilot views with one eye. The TV-like image is generated by the common module FLIR sensor integrated within the PNVS turret. The FLIR instantaneous FOV is 30x40 degrees but the pilot has the capability, through the Integrated Helmet and Display Sight System (IHADSS) to slew the turret-mounted FLIR in both azimuth and elevation to increase the field-of-regard (FOR). This concept approaches the normal daylight "out-of-window" flying by allowing the pilot to see FLIR imagery of the scene in the direction he turns his head. The high resolution FLIR imagery allow

the pilot to execute low level, contour, and nap-of-the-earth (NOE) flight to enhance survivability. In addition, the PNVS allow a night confined terminal area capability to execute maneuvers such as hover, bob-up, remask, sideward flight, takeoff, landing, etc., with confidence.

SURVIVABILITY

Since the Apache is an armed combat helicopter intended to operate in a high threat combat environment, it was designed to be extremely survivable.

It is tolerant to 7.62mm machine gun fire and 23mm cannon fire damage and continues to fly for enough time to egress the battle zone and return the crew to a safe area. For example, the main transmission and gearboxes are designed to operate for one hour without lubrication. The main rotor blades can withstand 23mm hits and continue to perform Survivability was given a great deal of attention in the engineering design phase. The vulnerability of this helicopter has been greatly reduced by reducing detectability through the use of low flicker main rotor blades, composite materials, low noise signature, glint-reducing flat transparencies, a passive engine exhaust infra-red suppressor and an infra-red jammer. When detected by hostile weapons systems the Apache is capable of taking evasive action up to +3.5 g's and -0.5 g's to avoid enemy fire. The aircraft is extremely crashworthy. The design ensures that the crew can survive a vertical impact with the ground at up to 42 feet per second or approximately 35 mph. The rugged construction and innovative design features that contribute greatly to an expected low attrition rate also help to assure that both crew and helicopter can reenter combat, promoting overall fleet survivability and life cycle cost savings. (Figure 4).



Crashworthiness

FIGURE 4

The Aircraft Survivability Equipment (ASE) suite consists of a passive radar warning receiver, IR jammer, chaff dispensers, and a radar jammer and laser detector. The ASE enables the Apache to stand and fight while rendering threat systems ineffective.

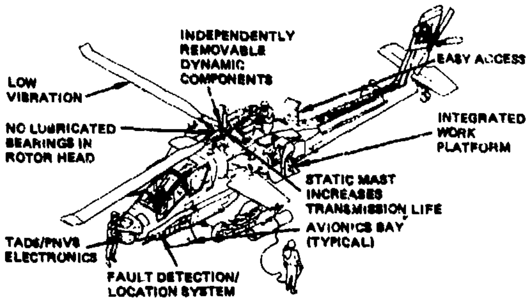
MAINTAINABLE

The Apache was designed with the capability to be maintained by the front line Army field mechanic at a lower maintenance hour per flight hour ratio than any previous attack helicopter. During operational testing, three YAH-64s achieved less than six hours of maintenance manhours per each flight hour (mmh/fhr), which was significantly better than the Army requirement of eight-13 mmh/fhr. Production Apaches are continuing to achieve six mmh/fhr. The features that make the Apache a highly reliable and maintainable helicopter are: an on-board fault detection/location system that continuously monitors critical components; build-in work platforms; easy accessibility to all compartments and bays for quick removal and replacement of components and black boxes; use of quick disconnects for electrical, hydraulic and fuel lines; engineer removal in 30 minutes; and a static main rotor mast which allows transmission removal without disturbing the main rotor head system.

TRANSPORTABLE

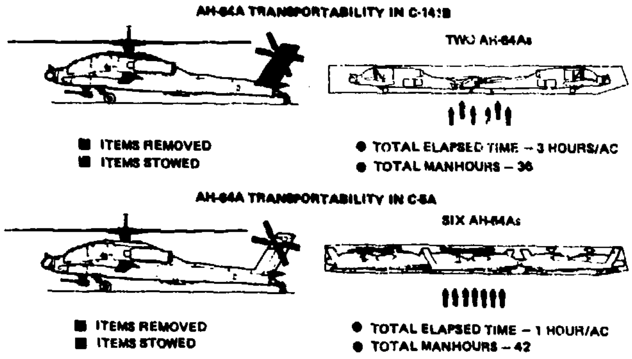
The AH-64A is capable of rapid strategic deployment worldwide and is air transportable in C-130, C-141 and C-5 aircraft. One Apache can be carried in a C-130, two in a C-141 and six in the C-5A. Air transport preparation time varies with the lift aircraft involved. For maximum transportability the main rotor blades, rotor mast and hub, tail rotor, stabilator, weapons platforms and 30mm weapon, may be readily removed. Six Apaches may be prepared for C-5A transport in 6 hours. All the preparations for air transport can be accomplished within the Army's elapsed time requirements. Preparation for flight is a simple reversal of the process (Figure 6).

Note the folded main rotor blades, and empennage as well as the kneeling landing gear can reduce the preparation time required.



Easy Maintainability

FIGURE 5



Transportability

FIGURE 6

SELF-DEPLOYABLE

Auxiliary fuel tanks provide self-deployment for the Apache for an 800+ nautical mile ferry range in a 20-knot headwind with a 45-minute fuel reserve at maximum flight speeds. Within a theater of operation, the Apache is easily deployable on internal fuel only. This capability is a fallout from the demanding endurance requirements of the combat missions.

CHANGING REQUIREMENTS DRIVE SYSTEM EVOLUTION

The preceding paragraphs summarize how an operational requirement was translated into an advanced weapon system through the application of the most advanced technology available. In recent years, significant changes have occurred in the U.S. Army's battle doctrine, in the Soviet threat and in the technologies available to the NATO community to implement new doctrine and counter new threats. McDonnell Douglas Helicopter Company completed an evaluation of these changes and has compiled a list of technology candidates for consideration by the U.S. Army for incorporation on the AH-64 of the future.

This study began as an investigation of all possible technologies that could be potential candidates for an advanced AH-64 Apache. These improvements were compiled from industry and government agencies and are based on preliminary concepts of the perceived threats anticipated through the year 2000. Additionally, a review of the U.S. Army RDT&E Plans was conducted to assure synergism with other growth programs and technical developments.

Once a preliminary list of potential configuration candidates was developed and screened by McDonnell Douglas Helicopter Company, U.S. Army representatives from the Apache Program Office, U.S. Army Aviation Center, the Army technical community and the U.S. Army Aviation Systems Command assisted in the selection of technologies that merited further study and consideration. At this point each technology was evaluated to determine the approximate nonrecurring cost and schedule required to qualify the technology change on the Apache. This allowed McDonnell Douglas Helicopter Company and U.S. Army planners greater insight and judgement towards establishing an advanced configuration that defeats the threat in the most effective manner for the least cost.

The criteria established for effectiveness analysis based on the Army 21 doctrine include:

a. Extended battlefield: strike deep, fight isolated.

b. See deep: viable intelligence, command and control.

c. Prepare to fight anywhere: self-deployment to any of the world's hot spots.

d. Mission Sufficiency: austere logistics, austere manpower, degraded modes.

e. Decentralized execution: new organization, flight isolated.

f. Real time acquisition/prioritization of targets.

g. Unity of effort: viable command and control.

h. Quick execution: high mobility, turn inside enemy's decision cycle.

i. Momentum of the attack: lethality/effectiveness vs. numbers.

j. Seize and retain initiative: an offensive strategy.

k. Separate enemy forces: fight isolated.

l. Force enemy to change dispositions: disrupt enemy battle plan at critical time.

m. Keep enemy off balance: offensive tactics.

n. Continuous operations.

TECHNOLOGY UPDATES RECOMMENDED

Preliminary design level assessment conducted by McDonnell Douglas Helicopter Company in 1984 and 1985, resulted in the long list of original technology candidates being distilled to a meaningful recommendation for consideration by the U.S. Army. While all of the recommended candidates can be shown to contribute measurably to the expanded effectiveness criteria listed above, more detailed effectiveness and cost analysis are currently being conducted by the Army who will recommend a plan for future upgrades of the Apache configuration. Technologies recommended by McDonnell Douglas Helicopter Company and under study by the Army are listed below with a summary of their major attributes. The first priority, an advanced system architecture and crew station is the subject of a current engineering activity. As such, it is discussed in detail in the final section of this paper.

VISIONICS UPDATE

Attributes:

Improved search, acquisition and targeting capability.

Reduced mean time between failure.

Recommended configuration modification:

FLIR Digital Scan Converter.

Automatic Target Recognizer.

Image Auto Tracker.

NBC/ANVIS Compatible IHADSS.

Target acquisition/crewstation battlefield display.

COMPOSITE WET WING

Attributes:

Increases emergency self deployment range to over 1100 nautical miles.

Increases combat range without using weapons stores.

Reduces wing drag.

Recommended configuration modification:

Graphite/Epoxy construction.

Structural capability to carry 450 gallon external fuel tanks on inboard and outboard stores stations.

Include 40 gallon fuel cell internally in each wing.

AIR-TO-AIR DEFENSE

Attributes:

Provides Air-to-air fire and forget capability.

TADS, PNVS, IHADSS and the turreted gun enhance air to air effectiveness.

Recommended configuration modification:

Integrate air to air missile into Apache system.

Modify area weapon system for air to air.

CHEMICAL/BIOLOGICAL PROTECTION

Attributes:

Provide crew compartment filtration of particles smaller than sub micron size.

Provide for aircrew warning of CB contamination.

Allows Apache to operate and fight in contaminated areas.

Recommended configuration modification:

Install filters and sensors in the environmental control system.

Provide source for cooling the crew in protective clothing.

ADVANCED AIRCRAFT SURVIVABILITY EQUIPMENT (ASE) UPDATE

Attributes/enhancements over AH-64A:

Improved aircraft survivability.

Utilizes U.S. Army ASE developments.

Postures aircraft for advanced technology.

Incorporates ASE reaction computer to reduce pilot workload.

IMPROVED IR SUPPRESSOR (Figure 7)

Attributes/enhancements over AH-64A:

Self-cooled IRS ejection pumps.

Reduces existing IR signature 50 percent.

Reduces drag by 0.9 square feet over existing design.

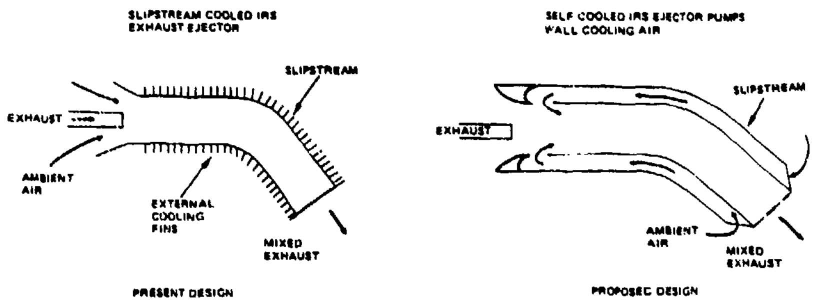


FIGURE 7 IMPROVED INFRARED SUPPRESSOR

ADVANCED FLIGHT CONTROL SYSTEM

Attributes/enhancements over AH-64A:

Reduces weight (140 pounds) and mechanical complexity.

Provides for dual redundant flight control paths.

Provides for auto pilot and auto hover capability.

Reduces ballistic vulnerability.

Improves handling qualities.

Utilizes technologies from existing qualified flight control systems.

Recommended configuration modification:

Eliminate all mechanical push-rods and bell cranks of the existing flight control system (Figure 8).

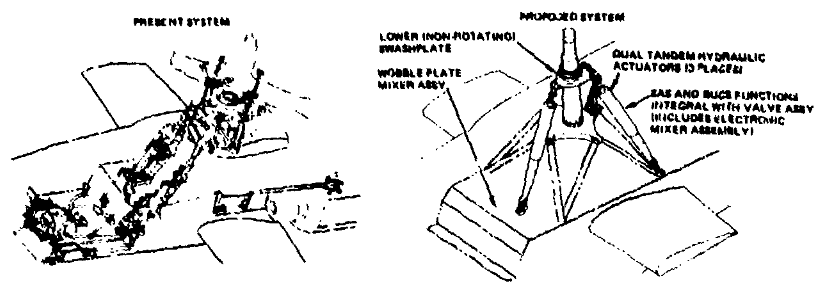


FIGURE 8 ADVANCED FLIGHT CONTROL

Replace the mechanical mixing unit with an electronic mixing unit.

Remove current cyclic flight controls and install sidestick flight controllers with electronic sensors and wiring.

Replace existing flight control actuators with direct-drive actuators (See Figure 11).

WIRE CUTTERS

Attributes/enhancements over AH-64A:

Increased margin of safety during Nap of the Earth flight during night or adverse weather conditions.

Recommended configuration modification:

See Figure 9

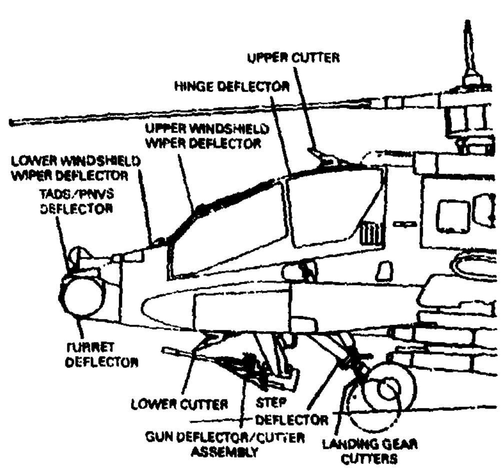


FIGURE 9 WIRE CUTTERS

CURRENT ACTIVITY AT MCDONNELL DOUGLAS HELICOPTER

While the U.S. Army is currently evaluating the relative importance of all of the technologies briefly described above, it is recognized that the area offering the largest payoff in terms of future growth capability and effectiveness enhancement is in the area of avionics, controls and displays. For that reason, an Independent Research and Development project was initiated by McDonnell Douglas Helicopter Company in July 1985 with a long term goal to develop and flight test an advanced technology avionics system and crew stations on an Apache. The objectives subsequently established for this new avionics and crewstation system are:

RELIABILITY. Increase system reliability by decreasing the LRU count, deleting any single point failures, and implement modern technology avionics and computer systems.

WORKLOAD. Decrease the workload in the crewstations by implementing a nonpaging automated control and display system developed and optimized in actual combat situations in our simulation facilities.

VULNERABILITY. Decrease vulnerability of mission critical avionics by redundant computing and sensing systems, judicious placement of components and embracing the distributed processing concepts for the avionics architecture.

OPERATIONAL AND SUPPORT COST. Decreases in operational support cost on the order of 50% for the avionics system. Cost of software maintenance and configuration control will be drastically reduced by standardizing stand-alone and in certain cases, embedded processors to MIL-STD-1750A Instruction Set Architecture. Programming will be done in compliance with the Department of Defense directive for the ADA High Order Language. Additional decrease in operational and support cost can be realized through the implementation of reliability initiatives such as reduction of LRU count, expansion of the on board fault isolation and recording system, and implementation of the two level maintenance concept.

CAPABILITIES. Provide ease of growth to accommodate new requirements in the area of weaponization, navigation, communication and target acquisition.

DEVELOPMENT METHODOLOGY

The engineering design and development methodology implemented for this project consists of four distinct but interactive phases necessary for a top down design approach in a software sensitive environment. (Figure 10)

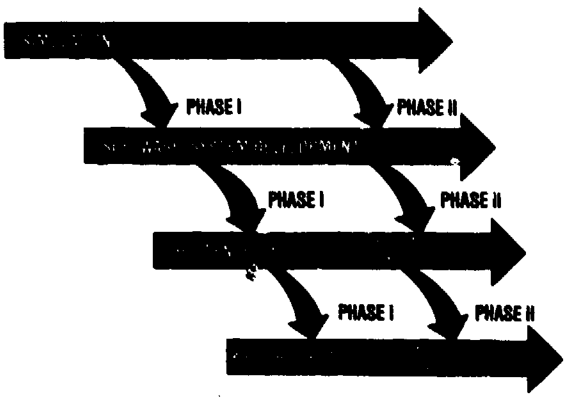


FIGURE 10

SIMULATION

The simulation phase develops and optimizes the basic man-machine interfaces. A broad brush task analysis and system operating requirements are translated into a baseline control and display configuration in the simulator. Full mission scenarios are then run to define overall system requirements, with part task emphasis on high stress mission segments. The iterated products of these evaluations (including crewstation configuration, MFD formats/symbology and switchology), drive the design of both the crewstation and the associated avionics system architecture. The resulting interdependent nature of the crewstation equipment and avionics systems permits a high degree of subsystem monitering and mission operations to be automated with an efficient priority interrupt action in the crew station. After the initial crewstation system design is defined and passed on to a "flight system" development activity, the simulation model is refined. Additionally, as the flight system is developed and refined, changes driven by "real world" implementation are traded with the existing model, thereby using the simulator's empirical output as a key element throughout the design process. As the design evolves in the next phases, the simulator assumes an increasing role in verifying and implementing the Army's tactics and doctrine in a continuing refinement of the man-machine interface.

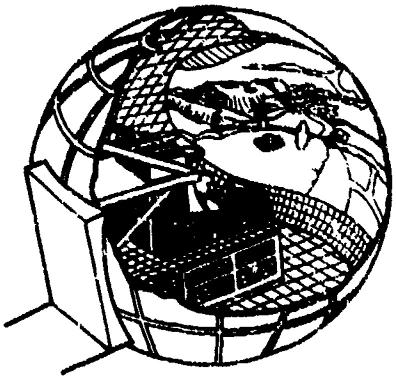


FIGURE 11 SIMULATOR

SOFTWARE AND HARDWARE DEVELOPMENT.

CONTROL AND DISPLAY

Figure 12 is a pictorial representation of the existing Apache A rear crewstation (pilot) where control and displays were implemented in a relatively autonomous fashion.

FIGURE 12 AH-64A PILOT CREWSTATION

The dramatic decrease in crewstation clutter is evident in Figure 13 which shows the advanced Apache crewstation. The primary displays are two 6 by 6 inch, shadow mask, full color Multifunction Displays; and an improved Helmet Mounted Display System. A 2 by 6 inch electroluminescent full color graphic is used to show status, caution warning and full emergency backup instrumentation.

Control is provided by four independent methods: A full alpha numeric keyboard with a single display line for visual feedback, touch screens overlayed on the Multifunction Displays, a cursor control, and eventually a voice interactive system.

Additional workload reduction is provided by a Data Transfer system which will preload all communication frequencies, navigation and target waypoints, and other mission planning data. The same system will also record for retrieval information throughout the mission such as caution warnings, LRU failures, new target data, etc.

MISSION MANAGEMENT

The Control and Display System will manage the following functions:

- Communications
- Navigation
- Weaponization
- Target Acquisition and Designation
- Electrical power management
- Maintenance diagnostics and recording
- Electronic Warfare
- Flight Control System
- All housekeeping functions (i.e caution warning, checklists, fuel management, etc.)

FIGURE 13 ADVANCED APACHE PILOT CREWSTATION

AVIONICS ARCHITECTURE

Two MIL-STD-1553 time division multiplex busses are implemented as shown in Figure 14 to provide growth capability and high level of redundancy for reliability and survivability purposes. Two MIL-STD-1750A Mission Processors are used where one serves as a hot backup to the other. Weaponization control and management also resides in these two MIL-STD-1750A Processors.

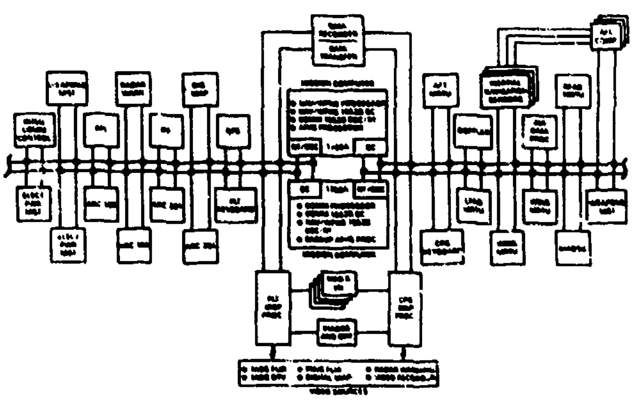


FIGURE 14 ADVANCED APACHE AVIONICS ARCHITECTURE

A unique feature of the architecture is the fact that software and hardware partitioning is implemented such that all system level processing is performed in the mission processors rather than at the sensor level. This approach allows for ease of technology insertion and growth to accommodate new or improved sensors. This approach also allows sensors to perform sensor level processing only, thereby decreasing sensor complexity and cost. Another important advantage is that all critical system level software is developed and implemented at the system integrator level.

McDonnell Douglas Helicopter Company has teamed with other manufacturers to develop the following hardware:

Sperry Flight Systems: multifunction displays, keyboards, MIL-STD-1750 display processors, LCD up front displays and various MIL-STD-1553B interface units.

Delco: MIL-STD-1750 mission computers

Leach Corporation: Electrical Power Management System

Canadian Marconi Corp.: Voice warning and audio distribution

Lear Siegler: Data transfer system and ground station

HOT BENCH

The Systems Integration Facility (Hot Bench) assembles and integrates the products of the previous phase. The equipment used in the Hot Bench is an evolutionary system which reflects the actual aircraft equipment as it become available (See Figure 15). Computers with preprogrammed flight paths will emulate external navigation sensors such as inertial reference systems and Doppler velocity radar to allow real time operation.

A real time testing and recording system will accurately validate all aspects of operational software envelope.

Changes to the system (hardware and software) are continuously fed back to the simulation and software development teams.

FIGURE 15 HOT BENCH

FLIGHT TEST

It is planned to modify a production AH-64A Apache to receive essentially a duplicate set of Hot Bench validated hardware for the purpose of flight tests and demonstrations. This aircraft will develop and assess the airvehicle integration facets of the advanced crewstation and avionics.

After the initial tests and demonstrations, this aircraft will continue to receive technology updates from a continuing McDonnell Douglas Helicopter Company independent research and development commitment for the Advanced Apache.

SCHEDULE

The implementation schedule of the crewstation and avionics development is as follows:

Simulation:

Initial control and display evaluation	completed
Advanced Apache CPG crewstation	Mid 1987
Advanced Apache Pilot crewstation	1988

System Validation Facility

Initial operation	Sept 1986
Advanced crewstation validated	1988

R & D Prototype Aircraft

First Flight, Advanced Crewstation	1989

CONCLUSION

Detailed planning for the Apache of the future is well under way and engineering work on the core of the advanced Apache - the system architecture - has begun. Every effort is being made to insure that this effort postures Apache to remain an effective weapon system well into the 21st century.

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(1) Brown, W.P., "AH-64A Apache - Battle Ready", Paper No. 21, Ninth European Rotorcraft Forum, Stressa, Italy, September 13-15, 1983.

SYSTEME DE MISSION SAR

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I PREAMBULE

Le nombre croissant de missions de recherche et de sauvetage en mer a conduit à rechercher un moyen de secours rapide, efficace et fiable.

L'hélicoptère, compte tenu de ses qualités de vol, est depuis longtemps utilisé pour ce type d'opération. Son système de mission, lorsqu'il existe est parfois rudimentaire et pas toujours parfaitement adapté.

La division hélicoptère de l'Aérospatiale a développé, en collaboration avec différents équipementiers (SFIM - BENDIX - CROUZET), un système spécialement adapté à la réalisation de ce type de mission, ceci grâce à :

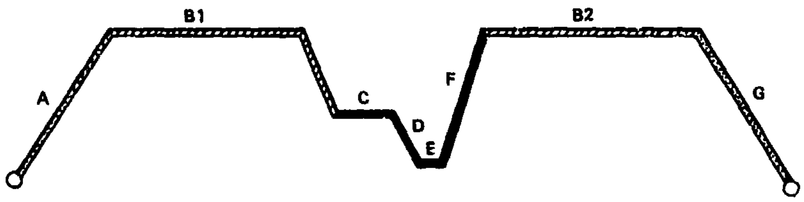
- un pilotage et un guidage entièrement automatique, particulièrement lors des trajectoires de recherche (PATTERN) et de la descente vers le vol stationnaire à proximité de la cible repérée,
- une planche de bord équipée de 5 E.F.I.S. (Electronic Flight Instrument System), écrans cathodiques pour l'affichage :
- des attitudes de l'appareil
- des paramètres de situation horizontales, sélectables en fonction de la phase du vol parmi 4 modes :
  - HSI
  - Secteur avec ou sans données RADAR
  - PATTERN de recherche
  - Stationnaire (HOVER)
- des données du radar de recherche sur un écran plus spécialement destiné à cette fonction.

LISTE DES SYMBOLES

AFCS	:	Automatic Flight Control System (Contrôle automatique du vol)
ALT	:	Altitude barométrique
A/S	:	Air speed (Vitesse Air)
BATIE	:	Boitier d'Adaptation et de traitement des informations d'entrée
BGS	:	Boîtier générateur de symboles
CDV	:	Coupleur directeur de vol
CR.HT	:	Cruise Height (Altitude radio de croisière)
DID	:	Dispositif d'insertion de données
EADI	:	Electronic Attitude Display Indicator (Indicateur d'attitude électronique)
EFIS	:	Electronic Flight Instrument System (Système d'Instrument de vol électronique)
EHSI	:	Electronic Horizontal Situation Indicator (Indicateur de situation horizontale électronique)
F.UP	:	Flight-up (Vol vers le haut)
G.SPD	:	Ground Speed (Vitesse sol)
HDG	:	Heading (Cap)
HOV	:	HOVER (Stationnaire)
H.HT	:	Hover Height (Hauteur radio de stationnaire)
ILS	:	Instrument Landing System (Système d'atterrissage aux instruments)
NAV	:	Navigation
PTN	:	PATTERN (Circuit de recherche)
RDR	:	RADAR
RMI	:	Radio Magnetic Indicator (Indicateur de cap magnétique)
SAR	:	Search and Rescue (Recherche et Sauvetage)
SCT	:	Secteur
T.DWN	:	Transition Down (Transition vers le bas)
T.UP	:	Transition Up (Transition vers le haut)
VOR.A	:	VOR approche
V/S	:	Vertical Speed (Vitesse verticale)
WPT	:	WAYPOINT (Point tournant)

2 INTRODUCTION

On peut décrire une mission SAR à partir d'un schéma représentant le profil du vol.



Ce profil se décompose en huit (8) phases :

- phase A : décollage+montée à l'altitude de croisière
- phase B_1 : rejointe de la zone de recherche
- phase C : trajectoires de recherche (Pattern de recherche)
- phase D : descente vers le stationnaire (Transition down)
- phase E : stationnaire (HOVER). Opérations de treuillage
- phase F : montée vers l'altitude de retour (Transition UP)
- phase B_2 : croisière retour
- phase G : descente - approche - atterrissage

Les phases A - B_1 - B_2 - G sont des phases classiques du vol. Les phases C - D - E - F correspondent aux phases de la mission SAR proprement dite.

phase C — L'hélicoptère est arrivé sur la zone de recherche. Dans cette phase il devra couvrir cette zone jusqu'à localisation de son objectif.

phase D — Cette phase consiste à effectuer une transition vers le bas pour venir se placer en stationnaire au-dessus de l'objectif.

phase E — Maintien du stationnaire jusqu'à la fin de l'opération de treuillage.

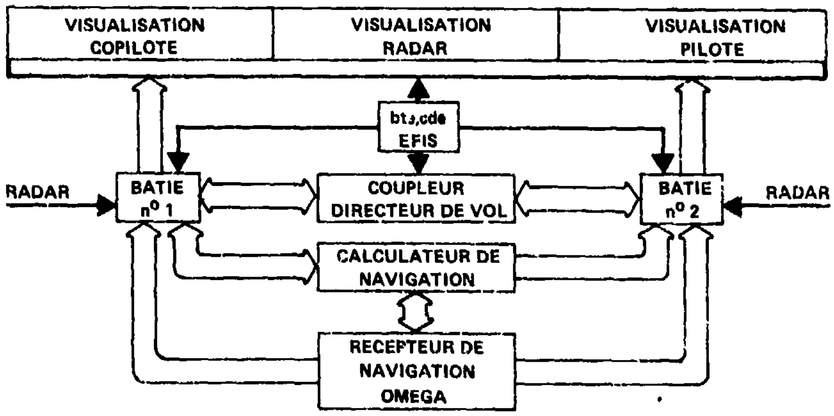
phase F — Transition vers le haut pour rejoindre l'altitude de croisière retour.

3 SYSTEME DE MISSION

Ce système a été conçu pour satisfaire à l'ensemble des phases de la mission et plus particulièrement aux quatre phases principales ci-dessus. C'est pourquoi sa description porte principalement sur les possibilités qu'il offre pour leur exécution.

Bien que l'ensemble soit parfaitement intégré, pour clarifier cet exposé, le système de mission a été scindé en trois parties :

- un système de navigation
- un système pilotage automatique (AFCS)
- un système visualisation et radar.



3.1 Le système de navigation

Le système de navigation est articulé autour de deux calculateurs. Un calculateur principal qui assure la totalité de la gestion navigation et un récepteur OMEGA qui en fonctionnement normal a un rôle d'équipement périphérique.

En cas de défaillance du calculateur principal, le récepteur OMEGA retrouve sa fonction calculateur de navigation plus récepteur OMEGA et assure automatiquement la poursuite de la navigation en cours ou d'une route programmée.

Le calculateur principal élabore trois positions présentes issues de trois modes de navigation :

– navigation DOPPLER
– navigation VOR/DME
– navigation OMEGA

Ces trois positions sont fournies en permanence, le choix du mode de navigation est fait par l'équipage en fonction de la phase de la mission.
Lorsque ce choix est effectué, l'équipage sélecte le type de route de navigation. Six possibilités lui sont offertes :

Navitation FROM - TO

Navigation sur une route directe entre un point FROM et un point TO.

Navigation Direct TO

Ce type de navigation peut se dérouler selon deux processus :

- **sans radiale**, ce qui correspond à se dérouter de sa position présente vers un but TO.
- **avec radiale**, ce qui correspond à se dérouter de sa position présente pour soit rejoindre un but TO ou s'éloigner d'un but FROM selon une radiale choisie par l'équipage.

Ralliement d'un but mobile

A partir des caractéristiques du but mobile (vitesse et route), des conditions présentes du vol, (vitesse et vent), le calculateur détermine un but de ralliement et effectue une navigation Direct TO vers ce but.

Navigation sur une route

Ce calculateur a la capacité de conserver en mémoire 10 routes de navigation pouvant comprendre jusqu'à 10 points tournants (WAY POINTS) sélectionnés parmi l'ensemble des points mémorisés dans le calculateur.

L'ensemble de ces points et de ces routes peut être inséré dans le calculateur au moyen d'un dispositif d'insertion de données D.I.D. (Data insertion device) ou manuellement.

PATTERN de recherche

Parmi les phases importantes de la mission SAR, la phase recherche (phase C) a une place privilégiée, car de sa réussite dépend le succès de la mission.

Pour cela, l'équipage dispose de trois circuits de recherche (Pattern) :

– Pattern échelle (creeping ladder)
– Pattern carré expansé (expanding square)
– Pattern secteur (clover leaf)

Mise en stationnaire (HOVER) - PATTERN APPROACH TO HOVER

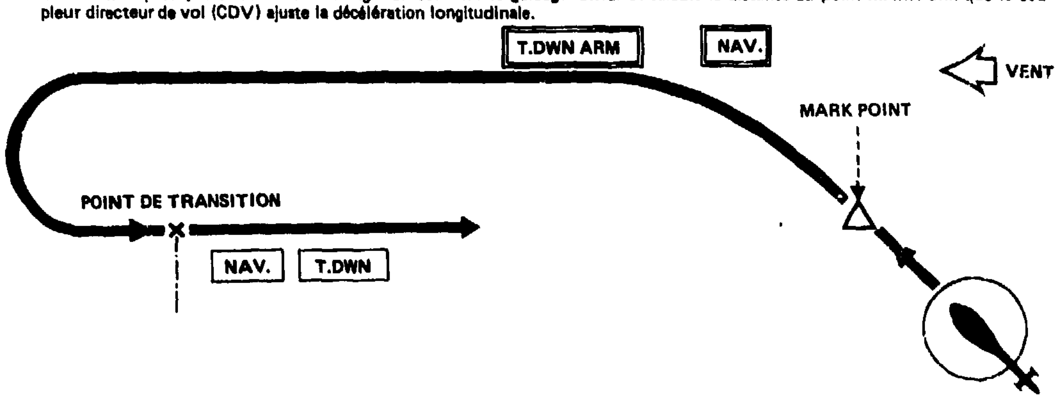
Cette navigation a pour but d'amener l'hélicoptère à une mise en stationnaire face au vent (phase D).

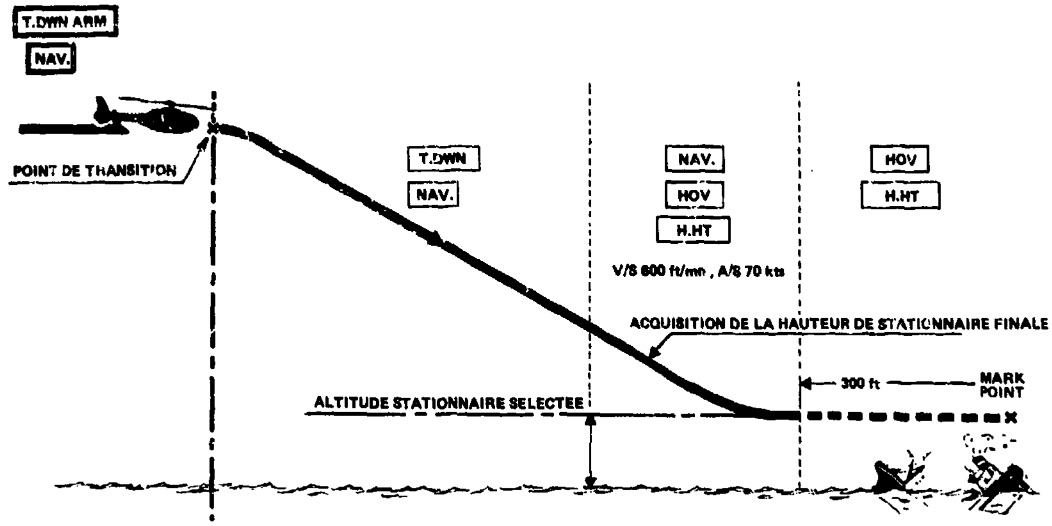
Le point de stationnaire (MARK POINT) peut être, soit un but désigné sur l'écran radar par l'intermédiaire d'un mini-manche (JOY STICK), soit un but défini au passage de sa verticale en «fixant» sa position sur le poste de commande du calculateur de navigation.

Cette navigation s'effectue en deux phases :

– la première amène l'hélicoptère face au vent sur le point de transition (TURN POINT)
– la deuxième phase consiste en une transition vers le bas du point de transition vers le point MARK suivant une direction fixe.

Pendant la première phase, le calculateur fournit un roulis commandé (ROLL STEERING) au coupleur directeur de vol, pendant la deuxième phase, le calculateur de navigation continue le guidage latéral et calcule la distance au point MARK afin que le coupleur directeur de vol (CDV) ajuste la décélération longitudinale.





3.2 Le système de pilotage automatique (A.F.C.S.)

Le système de pilotage automatique entrant dans la composition du système de mission SAR est un système 4 axes, constitué :

- d'un pilote automatique analogique duplex assurant la stabilisation de la machine autour des axes roulis-tangage et lacet
- d'un coupleur directeur de vol numérique bi-processeur assurant la réalisation sur les axes tangage-roulis et collectif des modes dits de croisière et des modes nécessaires à l'accomplissement des missions SAR.

Modes de croisière

Les modes de croisière disponibles au niveau coupleur directeur de vol sont les suivants :

ALT	:	capture et tenue d'une altitude barométrique
A/S	:	acquisition et tenue d'une vitesse air affichée
V/S	:	capture et tenue d'une vitesse verticale affichée
HDG	:	capture et tenue d'un cap affiché
NAV	:	en fonction de la sélection faite sur le poste de commande des visualisations : • interception et suivi d'une radiale VOR • interception et suivi d'une route issue du calculateur de navigation ou du récepteur OMEGA
ILS	:	capture et suivi d'un faisceau LOC et d'un faisceau GLIDE
VOR A	:	capture et suivi d'un faisceau VOR en approche
GO-AROUND	:	mode d'urgence correspondant à l'acquisition d'une vitesse air programmée = 75 Kts et d'une vitesse ascensionnelle V/S également programmée de 500 ft/mn.

Modes SAR

NAV	:	suivi du Pattern de recherche sélecté au niveau du calculateur de navigation
H.HT	:	acquisition et tenue en stationnaire d'une altitude radioélectrique présélectée entre 40 et 300 ft
CR.HT	:	acquisition et tenue d'une altitude de croisière radioélectrique présélectée entre 100 et 2500 ft
HOV	:	acquisition et tenue d'une vitesse doppler nulle
G.SPD	:	acquisition et tenue d'une vitesse doppler V_x et V_y
T.DWN	:	acquisition automatique d'une altitude radioélectrique présélectée et du stationnaire doppler
T.UP	:	transition à partir du stationnaire vers une altitude radioélectrique présélectée (CR.HT)
F.UP	:	mode de sécurité dans les phases de recherche et de stationnaire qui désengage automatiquement tous les autres modes dès que l'appareil descend en dessous d'une altitude radioélectrique présélectée et donne un ordre à monter avec une vitesse verticale programmée

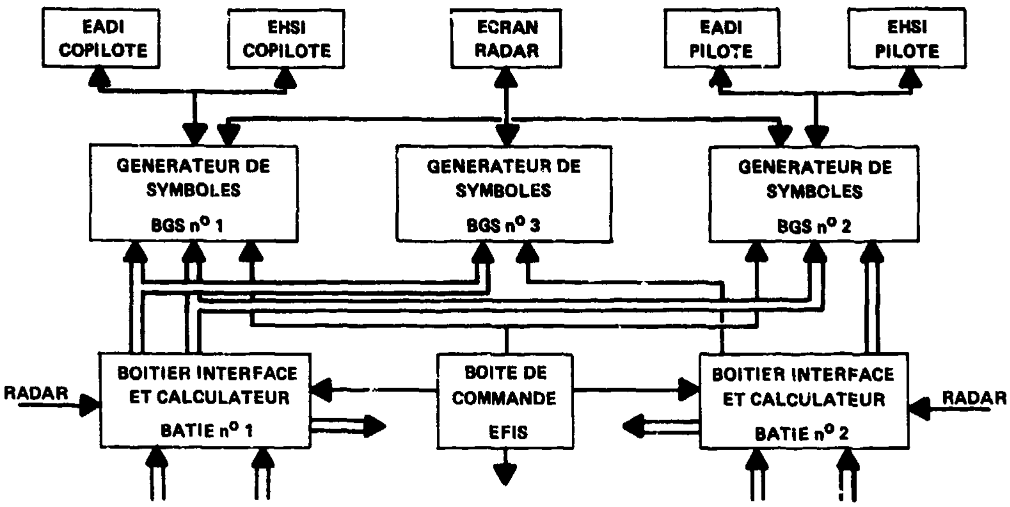
Exemple d'utilisation du coupleur

Phase recherche : les modes suivants sont généralement utilisés :

- CR.HT : Maintien de l'appareil à l'altitude radioélectrique présélectée. Ce mode est assuré par l'axe collectif.
- A/S : Maintien d'une vitesse air sur le pattern de recherche. Ce mode est assuré par l'axe tangage.
- NAV : Suivi de la trajectoire de recherche sélectée dans le calculateur de navigation. Ce mode est assuré par l'axe roulis.

3.3 Le système de visualisation et radar

3.3.1 Système de visualisation



C'est un système dual, bi-pilote composé de deux demi-systèmes identiques fonctionnant de façon indépendante et offrant une possibilité de reconfiguration en cas de panne détectée sur l'un d'eux.

Le demi-système n° 1 est affecté à la planche copilote, le demi-système n° 2 à la planche pilote.

Chaque demi-système se compose de :

- Deux tubes cathodiques couleur EFIS de type Shadowmask, installés sur la planche de bord et assurant les fonctions ADI et HSI.
- Un boîtier générateur de symboles (B.G.S.).
- Un boîtier d'interface (BATIE).
- Un poste de commande où sont regroupées les commandes nécessaires à la mise en oeuvre du système.

Le poste de commande permet :

– **Pour chaque demi-système**

- de sélectionner le mode de visualisation sur l'écran EHSI. La visualisation sur l'écran EADI est permanente.
- de choisir la source de navigation et de guidage.
- d'affecter une source d'informations à chaque aiguille RMI.
- de définir la valeur de la route à suivre (CRS)
- d'effectuer le test du demi-système.

– **Pour le système visualisation dans son ensemble**

- de définir la valeur du cap à suivre (HDG).
- de choisir l'échelle des distances du mode SECTEUR, SECTEUR + RADAR, PATTERN.
- de choisir le demi-système maître pour le coupleur.
- d'assurer la reconfiguration manuelle lorsqu'il y a mauvais fonctionnement d'un demi-système.

La visualisation générée par l'autre demi-système est alors envoyée sur le groupe d'écrans de celui en panne.

Description des commandes

1) Commandes séparées pilote et copilote

Ces commandes sont identiques entre elles. Elles comprennent :

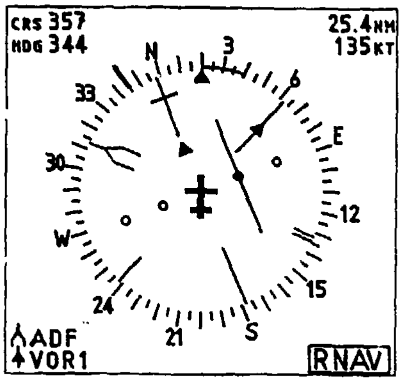
a) La sélection du mode de visualisation sur l'écran EHSI

Cette sélection est réalisée par appui sur l'une des huit touches fugitives dont six sont utilisées.

Les modes sélectables sont :

– HSI - SCT - RDR - PTN - HOV - ADI

La sélection est vérifiée par contrôle visuel sur l'écran correspondant.



Figuration HSI

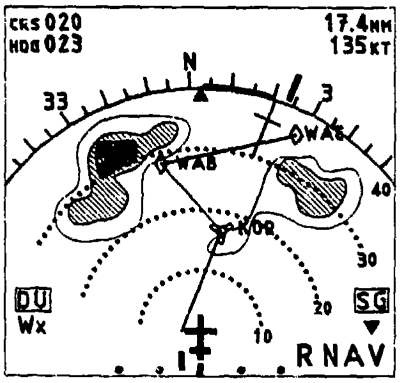
Cette figuration est semblable en ce qui concerne sa présentation générale à celle présentée sur un HSI+RMI conventionnel.

Elle comporte en plus les indications suivantes :

- distance balise ou prochain way-point
- vitesse sol
- route et cap sélectés
- source de navigation pour le HSI
- sources d'informations affectées aux aiguilles RMI.

Figuration secteur (SCT) et radar (RDR)

La figuration secteur représente la carte de navigation, fonction du mode de navigation choisi, dans le secteur avant de l'appareil. Les informations radar peuvent être superposées à cette figuration si la touche RDR est engagée.

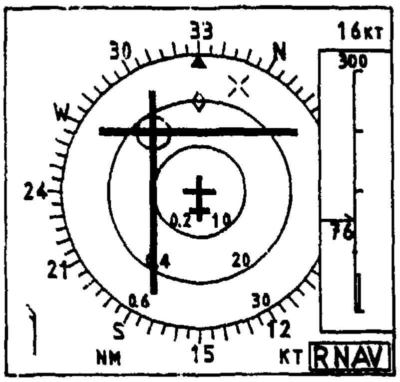


Figuration HOVER

Ce mode présente au pilote les informations nécessaires à la transition vers le stationnaire et à la tenue du stationnaire.

Les informations présentées sont les suivantes :

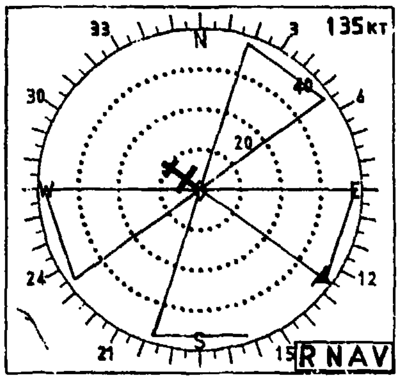
- cap magnétique appareil
- point de stationnaire
- vitesse doppler V_x et V_y
- vitesse sol
- hauteur radio-sonde actuelle et de consigne
- vent.



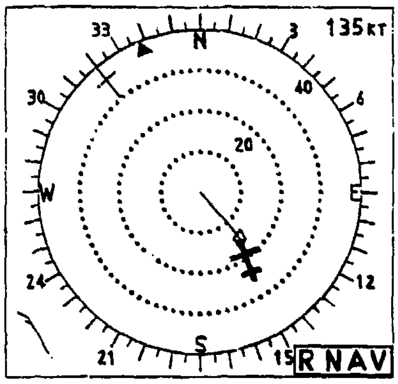
Figuration PATTERN

Ce mode présente le tracé d'un des «Pattern» de recherche mémorisé dans le calculateur de navigation. Cette figuration permet :

– soit une vérification visuelle des données entrées dans le calculateur de navigation lors de la préparation de la mission ou en vol lors d'éventuels changements des caractéristiques du Pattern.
– soit de suivre l'évolution de l'appareil sur un Pattern donné.

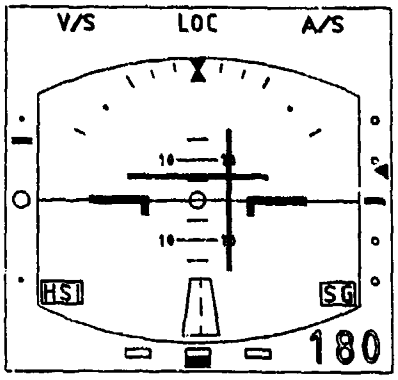


Ce mode de visualisation permet également la présentation du Pattern HOVER utilisé lors de la phase de vol Transition down guidée.



Figuration ADI

Cette figuration peut être appelée sur l'écran EHSI lors d'une panne de l'écran EADI. Elle reste identique à celle qui figurait précédemment sur l'écran EADI.



b) La sélection de la source de navigation

Chaque pilote dispose d'un sélecteur lui permettant de choisir la source de navigation dont il souhaite visualiser les informations.

Les sources sélectables sont :

R.NAV (Calculateur de Navigation)

Cette sélection permet la visualisation des informations issues du calculateur, fonction du type de navigation choisi sur son poste de commande. Ces informations sont visualisées en mode HSI-SECTEUR-PATTERN ou HOVER.

ONS (OMEGA)

En cas de panne du calculateur principal, l'utilisation du récepteur OMEGA se fait en utilisant la position ONS. Les informations issues de l'OMEGA peuvent être alors visualisées en mode HSI ou secteur.

VOR/LOC 1 - VOR/LOC 2

Visualisation des informations VOR ou VOR/DME en mode HSI ou secteur.

OBS

En mode secteur+radar permet à l'aide du bouton «CRS» l'affichage d'une droite orientée à partir de la maquette, droite visualisant la route future sélectée.

BCN

En mode secteur+radar, lorsque ce dernier est en mode Beacon permet à l'aide du bouton «CRS» l'affichage d'une droite orientée à partir du beacon sélecté. Cette droite visualise la route pour rejoindre le beacon.

TST

Position du sélecteur permettant d'initialiser le test du sous-système visualisation.

c) Affectation des deux aiguilles RMI

En mode HSI deux aiguilles RMI peuvent être visualisées sur le plateau de route HSI.

L'affectation de ces aiguilles aux moyens de navigation correspondant se fait à l'aide de deux inverseurs à trois positions :

- Inverseur 1 - aiguille fine
  VOR 1 - OFF - DF
- Inverseur 2 - aiguille double
  VOR 2 - OFF - ADF

d) Sélection de la route à suivre (CRS)

Une commande par bouton rotatif sans butée permet d'ajuster la route sélectée à la valeur choisie. Le contrôle se fait sur l'écran où un index matérialise cette route devant l'échelle des caps. En mode HSI et secteur, la valeur numérique de la route sélectée peut être lue dans le coin haut gauche de l'écran.

2) Commandes communes aux planches pilote et copilote

Le rôle des commandes communes est de contrôler des informations ou des fonctions qui ne peuvent prendre deux valeurs ou deux modes différents à un instant donné.

a) Sélection de l'échelle des distances NAV

Cette commande permet la sélection de l'échelle des distances en mode SCT, SCT+RDR, PATTERN. Sept échelles sont disponibles : 2, 5, 10, 20, 40, 80 et 160 Nm.

b) Sélection de la planche liée au CDV - Commande de reconfiguration du système

Réalisée par un commutateur à 4 positions, cette commande permet :

– En position N1 ou N2

- d'envoyer sur les écrans pilote et copilote des figurations indépendantes élaborées respectivement par le BGS 2 et le BGS 1.
- de lier le CDV à l'EHSI pilote ou copilote selon que l'on est sur N2 ou N1.

– En position S1

- d'envoyer sur l'écran pilote, lors d'une panne du BGS 2 ou du BATIE 2, la même figuration que celle présentée au copilote élaborée par le BGS 1.
- de lier le CDV à l'EHSI 1.

– En position S2

- d'envoyer sur l'écran copilote, lors d'une panne du BGS 1 ou du BATIE 1, la même figuration que celle présentée au pilote élaborée par le BGS 2.
- de lier le CDV à l'EHSI 2.

c) Commande de sélection de cap (HDG)

Une commande par bouton rotatif sans butée permet d'ajuster le cap à tenir à la valeur choisie. Le contrôle se fait sur l'écran où un index matérialise ce cap devant la rose des caps. En mode HSI et secteur la valeur numérique du cap sélecté peut être lue dans le coin haut gauche de l'écran.

3.3.2 Système Radar

Le Système Radar assure les fonctions météo et recherche. Il réalise les fonctions suivantes :

1. Modes météo

a) Mode WEATHER - WX

Les niveaux d'intensité des pluies sont représentés sur l'écran par des zones de couleurs suivantes :

– vert : de 1 mm à 4 mm/heure
– jaune : de 4 mm à 12 mm/heure
– rouge : plus de 12 mm/heure

b) Mode WEATHER ALERT - WXA

Même représentation que pour le mode WX mais la couleur rouge clignote.

De plus, si sur une distance de 45 NM au-delà de l'échelle sélectée, une information de niveau rouge apparaît, l'indication «TGT ALT» clignote en haut de l'écran.

2. Modes recherche

a) Mode SEARCH 1 - SRCH 1

Mode recherche utilisé au-dessus de la mer pour des échelles inférieures ou égales à 10 NM.

b) Mode SEARCH 2 - SRCH 2

Mode identique au précédent mais il ne possède pas la fonction «SEA CLUTTER REJECTION».

c) Mode SEARCH 3 - SRCH 3

Mode de recherche (maping) utilisable avec toutes les échelles.

3. Modes secondaires

En plus des modes primaires ci-dessus, le sous-système radar permet les modes secondaires ci-après :

a) LOG

Présentation sur l'écran radar de la liste des way-points et de leur coordonnées issues du système de navigation sélecté sur la boîte de commande EFIS copilote (soit R.NAV, soit ONS et appartenant à la route en cours).

b) CHECK LIST

Présentation sur l'écran radar de la check list pré et post flight.

c) NAV

Visualisation sur l'écran radar de la navigation choisie par le copilote sur sa boîte de commande EFIS (R.NAV, ONS, VOR/LOC 1, VOR/LOC 2). Cette information NAV peut être ou non surimposée aux modes primaires.

d) BEACON - BCN

- Visualisation des 2 impulsions BEACON et du code de la balise si celui-ci a été sélecté.
- Possibilité d'utiliser la fonction BEACON/TRACK comme sur les EHSI pilote et copilote.

Le mode BCN peut être surimposé aux modes primaires ou utilisé seul.

e) JOY STICK

Cette fonction permet la désignation d'un point de l'écran radar à l'aide d'un curseur que l'on peut déplacer en X et Y à l'aide d'un JOY STICK.

Ce point désigné est envoyé sur les écrans EFIS (mode SECTEUR-PATTERN-HOVER) et au calculateur de navigation.

Particularité du sous-système visualisation+radar

L'ensemble du sous-système visualisation+radar est conçu pour permettre la visualisation des informations :

- NAV ou NAV+RADAR côté pilote et copilote sur l'écran EHSI.
- RADAR ou RADAR+NAV sur l'écran radar installé dans la partie centrale de la planche de bord côté copilote.

Ceci implique la possibilité d'avoir deux échelles des distances. Une échelle des distances carte de navigation, une échelle des distances radar.

Cette capacité est obtenue par utilisation de deux interfaces radar (RIU1 - RIU2). L'une fonctionne avec l'échelle radar sélectée sur la boîte de commande radar, l'autre avec l'échelle sélectée sur la boîte de commande EFIS.

Ces deux interfaces travaillent alors selon deux modes :

- En mode normal

  S'il y a compatibilité entre les échelles, l'une des interfaces commande l'ensemble émetteur/récepteur et antenne et les deux interfaces exploitant le même signal vidéo issu de l'émetteur/récepteur.

- En mode «alternate Scan»

  S'il y a conflit entre les deux échelles , les deux interfaces se partagent par moitié le temps de balayage de l'antenne :

  - rotation de gauche à droite de l'antenne pour l'échelle 1,
  - rotation de droite à gauche pour l'échelle 2.

  Dans ce fonctionnement chaque interface exploite le signal vidéo issu de l'émetteur-récepteur lorsqu'il correspond à sa propre échelle.

  Le passage de l'un à l'autre des deux modes ci-dessus est automatique selon la sélection des échelles.

THE EH-101 INTEGRATED PROJECT: A NAVAL, UTILITY AND COMMERCIAL HELICOPTER SYSTEM

by

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SUMMARY

The Royal Navy and Marina Militare Italiana tasked EHI to develop a new shipborne helicopter.
The two Navies defined a common set of requirements related to the basic common helicopter plus a dedicated different set for each specific mission.
Concurrently EHI made an autonomous research on the potential EH 101 Civil and Utility Market. Through those analysis EHI came to the conclusion that a common basic helo could satisfy all the amalgamated requirements even if the basic roles were sometimes very different. Following is the exposition of the rationale and the trade off approach which was successfully devised and implemented to the aim of producing a multivariant A/C with the best compromise of a performance and cost.

1. REQUIREMENTS' ANALYSIS

In order to amalgamate the differing and competitive requirements for naval, civil and utility applications a comparative requirements analysis was carried out.
Tab. 1 gives a synthetic view of the more demanding requirements for the three basical roles : Naval, Civil transport, Utility.

TABLE 1

EH 101 INTEGRATED PROJECT OPERATIONAL REQUIREMENTS FOR NAVAL (R/N - MMI), CIVIL AND UTILITY APPLICATIONS

PERFORMANCE	NAVAL	CIVIL	UTILITY
Payload	-3000 Kg internal cargo -4500 Kg external cargo -20 people as secondary role	-30 passengers -5000 Kg internal cargo	-up 20 ÷ 35 troops -4500 Kg internal cargo -4500 Kg external cargo
Range/Endurance	4 HRS on station Roles ASW/ASV (Duncking sonar sonobuoy)	500 Nm (250 Nm radius) Passenger transp.	250 Nm Mixed transp.
Maximum speed	160 Kts	150 Kts	150 Kts
Cruise speed	140 Kts (S.L. ISA + 20°C)	150 Kts (3000 ft ISA+20°C)	150 Kts (6000 ft+15°C)
Hovering O.G.E.	SL ISA + 20°C and thrust margin 5%	- SL ISA + 30°C at max AUW - 6000 ft ISA+20°C at 95% max AUW	5000 ft ISA+20°C at max AUW
OEI performance	Flayaway max 100 ft SL isa+20°C at AUW	1000 ft runway Cat. A rejected take off	Complete Mission OEI

Max. AUW	13000 Kg	14290 Kg	14290 Kg
Agility	High	Normal	Normal
Weather cond. Temperature Icing Wind	All weather - 40°C + 50°C Heavy Take off and landing at 50 Kts ahead 35 Kts lateral 20 Kts behind	All weather - 40°C + 50°C Heavy NO specif.requir. except std FAA part 29	All weather - 40°C + 50°C Moderate NO specif.requir.
Blade folding	Yes	No	Yes
Tail folding	Yes	No	No
Landing system able to	± 3° pitch ± 8° roll	No specif. requir.	No specif.requir.
Landing gear	High energy	Normal	Crashworthy
Flotation	2 hrs at sea state 3	Sufficient to allow passenger to escape	2 hrs sea state 3
Helicopter in flight refueling (HIFR)	Yes	No	No
Crashworthy fuel system	Yes	No	Yes
Control	Single pilot	Single pilot IFR	Dual pilot
Cockpit panel	Advanced to reduce pilot work load Tactical presentation to pilot(s)	Advanced to reduce pilot work load	Advanced to reduce pilot work load.
AFCS	Able to incorporate special ASW/ASV model Mission survivability after first failure	Normal IFR single pilot	Normal
Navigation	Autonomous integrated high precision	Based upon ground support	Autonomous and based upon ground support
Avionic system	System monitoring and autoconfiguration capability Mission sensor integration/coordination.	System monitoring and autoconfiguration capability	System monitoring and autoconfiguration capability

Cost : For all variants to maximize commonality in order to lower non recurring and unit production cost.

Design priority	Maximize Availability and mission reliability	Maximize Safety	Maximize Availability

From the above table it is possible to highlight how different are the three sets of requirements and how each of them could lead to a complete different helicopter configuration.
Some examples will emphasize this statement.
In the naval role the payload has almost the same weight of the civil and utility ones, but the dimensions of the naval equipment are completely different from the passenger transport payload. Furthermore the need for a naval A/C to deploy a dipping sonar requires a great deal of mission time spent in hover condition. These requirements lead toward a cabin size narrower and long, while in a civil passenger transport role or in the mixed cargo-passenger transport a wider body fuselage is necessary. Figs 1, 2 shows the design process and the result is a solution that was driven by the civil/utility variant. But this solution would have direct

disadvantages upon the naval variant in term of weight, even if a greater flexibility is injected into the naval applications (see fig. 3).

In a similar way the requirement for agility it is mostly demanded by the naval variant in order to cope with the need to land in a confined space on ship, with strong gusty winds and a rough sea while for the Civil and Utility this is not required. The conclusion is that for an articulated elastomeric configured main rotor hub in a naval application the hinge offset would be greater while for the other variants 2.5% would be more than adequate (see fig. 4).
The choice for EH 101 was driven by the naval application and hence a 5% main rotor offset was selected. This choice of course has posed some penalties in terms of weight on the Civil/Utility variants (wider rotor head heavier configuration).
The same type of considerations were used for all the major requirements listed in table 1 and the majority of the helicopter features were designed in order to fulfil the requirements' envelope with consequences on weight.
Another aspect of the problem was the different design priorities assigned to the 3 basic helicopter variants.
Again in attempting to reconcile all the differing aims such as mission safety, reliability, helicopter availability, etc. EHI was obliged to envelop them in a sole set of requirements with direct consequences on the system design : obtaining a triple engine installation together with 3 hydraulic pumps, 3 electrical generators, etc.
The design choices were amalgamated in a such way as to minimize the weight impact, but nevertheless a weight penalty was recognized as unavoidable.
In such a situation the weight issue becomes the major design concern and the designers in order to achieve the optimum design were obliged to introduce as many new technologies features as possible (see fig. 5).
This process produced a good helicopter with a wider range of application (naval, civil, utility) and an all-up-weight (AUW) well within the trend of the existing in development helicopters (see fig. 6).

The result of this activity will be better exemplified in the next chapters but it is possible summarize it in the following : the new technology approach was successfully applied through :

- Wide use of composites (main and tail rotor heads and blades, tail unit, superstructure) (see fig. 8)
- Split load path technique also in main and tail rotor head.
- Wide utilization of on-board computing for aircraft management, data processing and cockpit presentation.
- Health and Usage Monitoring, through the aircraft management computer, for transmission dynamic system, structures and systems.
- Integrated avionic architecture
- All CRT cockpit displays

2. REQUIREMENTS VS. DESIGN FEATURES

The preceding chapter illustrated the general requirements for the three basic variants of the EH 101.
In order to define a configuration these are translated into detail design requirements at subsystem level. The process of the actual definition is actually very complex and requires both quantitative and qualitative analysis. Design requirements as a general rule are conflicting with one another, and this is more so when 3 basically different missions are to be fulfilled by one aircraft.
The optimization process is first performed at high level where basic design philosophies are stated. These are then translated into preliminary design. At this stage quantitative optimization is possible and directs the iteration process until the best compromise in line with the design goals is reached.
A top down design approach indicates as basic goals for the EH 101:

- Safety, for the civil missions.
- Mission reliability, for the naval/military.
- Extended performance and operational envelope for all roles.
- Improved life cycle cost for all roles.

These together with basic aeronautical design philosophies (foremost is weight minimization) and a Production and Market oriented approach have driven Development effort along the following guidelines:

- Maximum commonality
- Modular design with interchangeability features where commonality is not cost effective.
- Extensive use of new technologies, in materials, avionics, aerodynamics and design.
- Redundancy in structural and system design.

- On board computerised A/C Management System.
  Health and Usage monitoring of Dynamic components structure and systems
- R & M oriented design.
- Maturity programme in advance of the entry into service of the production helicopters.

The following paragraphs show how operational requirements were translated into design requirements and then into subsystem configuration.
The result at system level is discussed in the conclusions.
The EH 101 configuration appears to be a step improvement in safety and performance with a competitive empty weight/AUW ratio (see fig. 6).
The costs also (fig. 7), in terms of manufacturing hours, shows a satisfactory allocation, taking in configuration the relatively severe requirements to be met.

2.1. MAIN ROTOR

2.1.1. DESIGN REQUIREMENTS

DRIVING REQUIREMENTS	CONSEQUENT DESIGN FEATURES
Max AUW : 14290 Kg (Civil) 13000 Kg plus growth (Naval)	M/R blades and hub sizing
Power folding	- Tension link with blade fold system
Wind on Ground : 65 KTS parked 60 KTS during folding and reeving up and down	- High stiffness design of M/R hub, tension link and blades
Foldability of A/C M/R and Tail Unit within 52 x 18 x 17 ft box	- Folding hinges at a high radial station
Maneuverability in flight and during take off and landing on ships with gusty winds 50 KTS up ahead, 35 KTS , lateral 20 KTS behind	- Flap hinge offset at 5% R
Low vibration levels in cockpit and cabin at Cruise speed (.05 g in the cockpit)	- Low hinge offset - 5 blades (0.1 g) - Head absorber option
Operation in full icing conditions at cruise speed - 10 KTS, 10000 ft, - 30°C	Deiced blades through electrical heater mats
Maintainability (reduced MMH/FH)	- Elastomeric bearings - Reduced number of parts - Electrical blade folding system instead of hydraulic.
Safety	
- Fail safe and damage tolerant features	- Dual load path for flap and lead/lag loads (composite hub/support cone) - Multiple loop windings for centrifugal loads (2 + 2 at each bearing) - Use of composite materials (low crack propagation) - Low stress in composite hub : sized by stiffness
- Resistance to lightning strike	Bonding
Weight	Maximization of composite Materials (carbon fibre and glass), Aluminum alloys, Titanium alloys

	Minimum size of M/R blades (chord and radius) compatible with performance (third generation distributed airfoils, Berp Tip)
Hover power	Minimum disc loading at Naval weight: (max. rotor diameter), minimum chord
Max flight speed	- Retreating blade stall : large blade area (chord, radius) or high maximum lift airfoils and BERP tip which delays entry into stall boundaries. - Advancing blade max divergence : Thin airfoils High divergence mach number airfoils - Swept (BERP) tip.
Max flight speed power	- Low blade area, low Cd airfoils, Low drag hub (Thin cross section, low radius, fairings)
Minimum External Noise	Low Tip speed, 5 blades
Quick rotor braking at high wind speeds and without shutting off engines	Lead lag Static/dynamic stops

2.1.2. Design Solution

The requirements are highly conflicting with one another. The best solution can be obtained only by a quantitative optimization exercise.
The resulting configuration is shown in figs. 9,10.
Its main features are :

- 5 blades 18.6 m. diameter, 0.68 m chord, 204 m/s tip speed, distributed RAE third generation airfoils, BERP tip.
- 5 elastomeric spherical bearings for centrifugal loads, plus 5 spherical centering elastomeric bearings for shear loads.
- Conventional dampers for initial A/C release. Elastomeric dampers are in development for later releases.
- All composite blade
- Mixed composite/metallic hub
- Composite 28% in weight
- Titanium 19% in weight
- Steel 38% in weight
- Al/Alloy 15% in weight
- De-iced blades
- Faired hub and tension links
- Two options are offered for the hub (The first is standard for naval applications).
  - Power folded blades
  - Non folding blades
- The weights are :
  - Folding rotor : 1470 Kg
  - Non folding rotor : 1290 Kg

Fig. 11 shows that the chosen rotor solution is on the general weight trend, while offering significantly higher performance and flexibility. At the specification naval weights (13000 Kg) it is slightly above the trend, but if weight growth is taken into account up to the civil A.U.W. (14290 Kg) the EH 101 M/R is about 80 Kg below the line.

If a non common non foldable configuration is chosen a significant improvement is obtained and the EH 101 configuration is 260 Kg below the trend line .
It is to be noted that for this particular issue the weight penalty of folding is significant thus commonality was not pursued.
Fig. 12 suggests similar comments in terms of manufacturing hours.

2.2. DRIVE SYSTEM AND POWERPLANT

2.2.1. Design Requirements

DRIVING REQUIREMENTS	CONSEQUENT DESIGN FEATURE
Total gear reduction 97.4 : 1	4 stages MGB
13000 Kg for Naval A/C	4640 HP T.O. rating
14290 Kg for Civil A/C	5100 HP T.O. rating
Hover at 35 Kts lateral wind with 15% pedal margin at 13000 Kg (naval AUW)	1170 HP Max power for T/R drive
Low specific fuel consumption	Engine operating at high power, thus having a low power margin above hover power.
Quick rotor braking : 15 secs from 80% RPM with engine in G.I.	- High performance rotor brake - 2 Actuated freewheels
Accessory drive on ground through one engine	Actuated freewheel on one engine switchable from main rotor mode to accessory power mode.
Maintainability	Accessibility to transmission refills' levels, pumps, filters, sensors. Ease of transmission assembly installation and removal. Accessibility and ease of removal of accessories (generators, hydraulic pumps).
Resistance to corrosive environment (sea operation)	No Magnesium Alloys.
Low weight/complexity	Twin engined solution
OEI PERFORMANCE :	
- Safe flyaway after engine failure at 13000 Kg, 100 ft above ground, SL ISA + 20°C, still air	High ratio of total contingency power O.E.I. relative to T.O. power. Ideally the ratio should be unity.
- AUW not limited by O.E.I. performance for Civil operations.	This is obtained through high engine power margins above hover power and/or a multi engined (3) configuration (allowing a lower power margin for each engine)
- Improvement in Lubrication system	- Separation of AGB and MGB. Oil loss in one does not affect safe operation. - Lubrication system completely duplicated in Main Gearbox : two sumps, two pumps, two circuits, two oil jets at each lubrication point.
- Oil loss capability 30 minutes at 120 Kts plus 2 Min at take off power	- Separation of AGB and MGB. Oil loss in one does not affect safe operation. - Dual lubrication system in MGB

	- Special configurations and materials for bearings.
	- Built in oil pockets inside the casing
- Early warning and diagnostic of failures	Health Monitoring System : - Oil pressure and temperature sensors - Bearing temperature sensors - "Intelligent chip detectors" (Quantitative Debris Monitoring). - Vibration Monitoring
- Minimisation of damage due to engine high energy rotors bursts - Reduction of vulnerability to bullet hits	Engine Separation

2.2.2. Design Solution

The resulting drive system arrangement is shown in fig. 13,14,15. Relevant characteristics of the drive system/transmission powerplant is :

- 3 T700 Engines : T700/401A for the Basic Naval Variant (1680 Hp each), 3 CT7.6A for the Civil Variant (2020 HP each) and for the Growth Naval Variants.
- Transmission designed for 4640 HP and matured to 5100 HP rating during Development
- Dual lubrication system
- Isolated accessory gearbox driving one 90 KVA and one 10 KVA generator, plus 2 hydraulic pumps.
- Main gearbox driving one 90 KVA generator plus one hydraulic pump.
- 3 Micron fine oil filtration to reduce gear and bearing wear
- Health Monitoring system (see fig. 16)

The resulting weight is shown in fig. 17, while manufacturing hours comparison is illustrated in fig. 18.

2.3 TAIL ROTOR

2.3.1. Design Requirements

DRIVING REQUIREMENTS	CONSEQUENT DESIGN FEATURES
4640 HP / 5100 HP T.O. rating	Hover thrust: 1500 Kg Min.
Hover with 50 KTS head wind 35 KTS lateral wind with 15% pedal margin at Naval weights	High pitch angles and high thrust capability
Fin blockage	High rotor diameter or small fin
Weight	Small tail rotor diameter and chord, and/or high lift coefficient airfoils. High tip speed.
Anticlockwise M/R rotation	Clockwise (looking from left of the A/C) T/R rotation so as to counteract M/R vorticity in lateral flight.
Clearance from ground	High Tail Rotor configuration
Low external noise	Low tip speed

Parking on deck with 65 Kts unrestrained	No flap/lag hinges
Simplicity / Maintainability	Bearingless configuration. Where hinges are required, use of elastomeric bearings.
Operation in ice conditions	Anti-iced blades
Safety	Damage tolerances : maximum use of composites

2.3.2. Design Solution

Fig. 19 shown the selected T/R configuration, whose main features are :

- 4 m diameter
- 0,305 chord
- 198 m/s tip speed

Clockwise rotation

Soft in plane all composite hub. Feathering is enabled through two elastomeric bearings per blade.
All composite blade with integral pitch horn.
Weight is shown in fig. 20
The particularly clean and simple design gives a satisfactory cost comparison as indicated in fig. 21.

2.4 AIRFRAME AND TAIL UNIT

DRIVING REQUIREMENTS	CONSEQUENT DESIGN FEATURES
- Loading capability for : 30 passengers 4500 Kg of cargo 3000 Kg of mission equipment	- Almost trapezoidal cabin section able to load 4 passengers for run - Standard floor rails for quick change of payload.
- Capability to carry a small vehicle inside the cabin	- Rear ramp
- Crashworthy airframe type	- Metallic type construction with high deformation frames. Landing gear provided with plasticizing interchangeable units.
- High ground ship deck mobility	- Wheeled, tricycle forward landing gear
- Inherent longitudinal stability to cope with the tail rotor loss	- Vertical fin
- Structural redundancies in the main frames.	- Fail safe/damage tolerant design, large use of composites materials.
- Heavy icing condition	- Anti-iced horizontal stabilizer, engine intakes and windscreen
- Easy loading of cargo and passengers	- Wide (2 m) starboard door. For cargo Airstair port door for passengers and crew. Rear ramp door for small vehicles and large cargos.

2.4.1. Airframe Design Solution

In order to comply with all the above requirements the choice was to design a modular airframe because the requirements were too different to each other and the weight penalty associated unacceptable for a single airframe.
The four modules are : (see fig. 22) forward fuselage, cabin, rear fuselage and tail unit.

The forward fuselage module is built in two different technologies: the lower side is traditional metal construction with frames, bulkhead and metal panels, while the upper glazing structure is completely composite construction (see fig. 25).
The main cabin is simply built in frames and honeycomb panels with two main beams crossing the frames in the roof area to transfer the main gearbox attachments' loads (see fig. 26).
The reason why the lower forward fuselage and the main cabin is in metal construction is mainly due to the crashworthy requirements.
The third module is the rear fuselage where the 3 variants have their major differences : each variant has a dedicated rear fuselage (see fig. 22) one for naval, one for civil comprising a wide baggage compartment, and one for the utility with a rear ramp. Rear fuselage type of construction is conventional semimonocoque and it is connected to the cabin and to the tail unit through modular joints.
The last airframe module is the tail unit comprising vertical fin, horizontal stabiliser, intermediate and 90° gear box attachments.
The type of construction is completely in composite (see fig. 27) and this reduced the weight, in respect to the conventional construction, and also cost because of the simplicity achieved is such that all the complete unit is manufactured in only 20 pieces.
While the tail unit will be initially safe life certified the type of construction is such as to have an inherent fail-safe capability that will be utilized in a later stage to meet a damage tolerant certification. Finally all the superstructure for engines and transmission cowlings, compartments and fairings are basically in composites material.(see figure 28)

Analysis of the diagrams in fig. 23 and 24 shows that also in this case the weight and costs of the cabin are almost in line with trends.

2.5.1. Control System and Hydraulics Design Solution

The control system is a conventional mechanical low friction design with push-pull rods and bellcranks : very simple and reliable.
The mechanical system is directly connected to the servo valves on the 3 hydraulic jacks controlling a conventional fixed swashing plate. A similar system connects the pedals to the tail rotor hydraulic jack. Cyclic-collective and collective yaw are also connected through a mechanical interlink (see fig. 29).

The hydraulic system is described in fig. 30 with the 3 main integrated hydraulic pumps connected with two change-over solenoid valves and a fourth electrically driven AC pump for ground checks and emergency brake accumulator recharge.
Two hydraulics pumps are installed on the auxiliary gear box, while the third one is installed directly on the main gear box. In such a way in case of complete failure of the auxiliary gear box, at least until the main rotor is running there will be hydraulic power to control it . (see fig. 13)

2.6 ELECTRICAL SYSTEM

2.6.1. Design Requirements

DRIVING REQUIREMENTS	CONSEQUENT DESIGN FEATURES
Power for avionics including radar and sonar, blade fold, engine intake and windscreen anti-icing	45 KVA
Additional power for rotors and tail plane anti-de/icing	90 KVA
Full capability after generator failure	2 main generators
Emergency power after 2 generators failure	Additional 10 KVA emergency generator
Independency of generator failures	1 main generators installed on the main gearbox, plus one main generator on the Accessory Gearbox

2.6.2. Design Solutions

The selected electrical configuration reflects the features seen above (see fig. 31)
Commonality is not cost effective as regards anti de-icing capabilities thus interchangeable 45 KVA or 90 KVA generators are fitted, plus a common 10 KVA generator.

2.7 AVIONIC SYSTEM AND DYSPLAYS

The operational requirements for this system are different for all civil, naval or utility variants. A strong degree of commonality was certainly not achievable. On the contrary the safety issue, considerable pilot work load reduction and a synthetic presentation of data were common and strong requirements.

DESIGN REQUIREMENTS	CONSEQUENT DESIGN FEATURES
- Reduction in pilot work-load : - Data presentation - System redundancy and automatic reconfiguration after the first system failure	- C.R.T. type cockpit displays and symbol generators. The inboard aircraft computer (AMC) as a key feature to control and manage the totality of the system's informations
- Data integration on the pilot's displays using the same display for different purposes (engine, navigation, warning, etc.)	- Computer, cockpit, sensors communication system reliable and integrated : Arinc 429 or MIL-STD-1553B
- Ease to adapt the cockpit information presentation to the mission requirements.	- Mission software dedicated and adaptable for any single usage.
- Systems and transmission health and usage monitored	- Specific H. & UM software built in the AMC.
- AFCS sensors segregation from monitoring sensors	- Physical separation between the AMC and the AFCS computer.
- Helicopter to be controlled after any single avionic failure.	- Stand-by instruments and display.
- Radar meteo informations integrated on the cockpit displays.	- C.R.T. radar presentation.
- Improve the cockpit information presentation.	- Coloured C.R.T.

2.7.1. Avionic System and Displays Design Solution

Fig. 32 shows the schematic avionic cockpit configuration for the naval variants. The architecture of the system is such that it allows to have 6 cockpit displays devoted to the aircraft management and 2 dedicated mission displays. This cockpit configuration is not the more suitable for a civil variant helo and for it a different configuration as in fig. 33 is foreseen.
Furthermore in order to better reconcile the two different solutions it is possible to arrive to a third configuration such as that illustrated on fig. 34.
All these solutions are possible thank to the great flexibility injected in the basic avionic system architecture hinged around the AMC. Fig. 35 and 36 show the two basic avionic architecture for civil and naval variants (AMS).
The main functions of the AMS are the following : Navigation, communications, performance calculation, sensor monitoring, check list, H & UM monitoring.
Segregated from the basic AMS there is another computing system dedicated to the autostabilization and autopilot (AFCS).
The autostabilization is a function common to all variant while again some mission peculiar functions will be different from one variant to the other (cable hover, etc.).
Analyzing the two basic avionic architecture schemes it is apparent that the basic items remain unchanged except for the communication media (MIL-STD-1553B versus ARINC 429) and related interface units.
The new technology approach (airborne computer and multimicro structure) allowed to obtain the benefits of " dedicated" solutions and at the same time to maintain a high degree of commonality between military and civil configurations with consequent development cost reduction, as better analized in paragraph 3.
Also the weight is contained within the conventional avionic system trend, but with considerable more capability. And finally the safety was greatly enhanced thanks to the duplication of all the main avionic systems (AMC, AFCS, cockpit displays, etc.) and the capability of the helo to be controlled also in case of complete integrated system black-out.

3. COST CONSIDERATION

3.1 COMMONALITY

At the current status of the program we can measure in terms of money the "commonality" of the three EH 101 variants.
As said before "commonality" was one of the main tasks and a philosophy driving all the possible choices during preliminary and detail design.
The following table gives the percentages of common parts cost over the total cost of the helicopter including bought outs and raw material (unit production cost at stabilized production).
Note that the naval variant shows a relatively low value of common parts. This is due to the large value of specific avionics and armaments.

EH 101 VARIANTS	COMMON PARTS VALUE	SPECIFIC MISSION PARTS VALUE
Naval	61%	39%
Utility	83%	17%
Civil	86%	14%

3.2 NON RECURRING COSTS (N.R.C.)

A comparison study has been conducted in order to measure the impact of extended commonality philosophy and of intercompanies cooperation on N.R.C.

- Table 2 synthetizes the results of this study on the EH 101 co-operative program.
- N.R.C.'s incurred by one company for a dedicated aircraft (translated to a EH 101 class helicopter), a three variants modular aircraft and a three variant modular aircraft designed with a second partner are compared.
- Lecture key of the table is column "A". Column "A" says that spending 100 $ for a dedicated helicopter project, 1 $ is spent for management, 30$ for engineering and so on. The other columns take column "A" as a reference. For example if we spend 1$ for management of a dedicated helicopter (column A), EH 101 program (column "D") will require 7 $ for management (addition of Agusta + WHL + EHI management costs)
- Column "B" indicates N.R.C. incurred by a single company in a three variants modular helicopter development.
Total N.R.C. would be 79% larger than the expenditure of column "A" reference case.
- Column "C" reports relative expenditures incurred by EHI-WHL-Agusta to develop EH 101 parts and systems common to the three variants.
- Column "C_1", "C_2", "C_3" give relative N.R.C. to develop add-ons equipments peculiar to each of the naval, civil and utility variants.
- Column "D" gives total (all variants) development program cost incurred by EHI + WHL + Agusta. Indication is given that all variants N.R.C. is 2.05 times the mono-role single company N.R.C.
- Columns "E_1", "E_2", "E_3" are giving N.R.C. incurred, by each company, to develop (up to stabilized production phase) the three main variants. More correctly speaking, figures represent cost incurred at nation level. Thanks to Government and National Financial Support, Agusta afforded cost are consistently lower, but these considerations are analyzed later.

TABLE 2 - IMPACT OF COMMONALITY AND WORKSHARING N.R.C.

EH 101 MAIN VARIANTS

	A	B	C	C_1	C_2	C_3	D	E_1	E_2	E_3
NON RECURRING COSTS ITEMS	REFERENCE COLUMN DEDICATED A/C DEVELOPED IN A SINGLE COMPANY COSTS %	INTEGRATED PROJECT 3 VARIANTS DEVELOPED BY SINGLE COMPANY	EH 101 COMMON %	EH 101 ADD-ONS NAVAL VARIANT %	EH 101 ADD-ONS CIVIL VARIANT %	EH 101 ADD-ONS UTILITY VARIANT %	EH 101 TOT.PROGR. $D = C_1+C_2+C_3$ %	EH 101 NAV IT.OR U.K. SHARE %	EH 101 CIV IT.OR U.K. SHARE %	EH 101 UT IT.OR U.K SHARE %
1) MANAGEMENT (including travels)	1	2,5	4	1,5	0,8	0,7	7	1,4	1,1	1,0
2) ENGINEERING - drawing & calcul. - specifications - tech.data coll.& reports - mock-ups	30	62	45	12	6	8	71	13,5	10,5	11,5
3) TESTING - ground - flight - prototyp.constr. - models - materials	39	75	58	15	4	8	85	7,2	11,7	13,7
4) TOOLING - including rigs	8	10,5	9,4	0,4	0,5	0,7	11	4,8	1,8	1,9
5) PRODUCT SUPPORT - including pubb. & R.& M. analisys	5	7	5	2	0,5	0,7	8,2	1,8	1,1	1,2
6) CAPITAL ASSETS - new machinery - new buildings	4	4	4	-	-	-	4	0,7	0,7	0,7
7) MATURITY PROGRAM	5	7	5	2	0,4	0,2	7,6	1,8	1,1	0,9
8) PRODUCT.LEARNING - up to 100th unit		11	8	2	0,9	0,5	11,4	2,3	1,8	1,6
TOTAL N.R.C.	100	179	138,4	34,9	13,1	18,8	205,2	40,5	29,8	32,5

NOTE : E_i values = $(C/3 + C_i)$ 1/2

We would like to add some comment on the values indicated on table 2, comparing mainly columns "D" (EH 101 full program) and "A" (dedicated, single company aircraft).

1) Management. This items shows a noticeable ratio of 7 (D/A. It is easy to understand why : intercompanies cooperation means meetings, discussions, travels, necessity of agreement on processes, methods, procedures and work-sharing, etc.

2) Engineering. Ratio D/A of this item is 71/30.
Reasons for this are the longer preliminary studies and the continuous evaluation and analisys of conceptual and detail solutions in order to meet the commonality task.
Distance between two companies designers also affects engineering work in terms of costs and time schedule.

3) Testing. Ratio D/A of this item is 85/39, mainly due to larger amount of prototypes to be evaluated and qualified. Any-how a significant number of tests and results will be shared and considered valid between the military and civil versions, if properly programmed and considered.
We can see that only 17,6% (= 15/85 . 100) of testing will be peculiar to naval variant, and 4,7% (= 4/85 . 100) to civil variant; while approximately 70% (= 58/85 . 100) of testing will be useful for the common helicopter, i.e. useful for all variants.

4) Tooling. Ratio D/A = 11/8 . Commonality has a beneficial affect on tooling cost, since only a small amount of tooling is dedicated to specific variants, while 85% of tooling is common. Note that, as a driving philosophy, pre-production or development tooling is designed and made as production tooling, and will be used in production phase.

5) & 7) Product Support; Maturity Program. The same comments apply here as for tooling; with approximately equal percentage of specific variant costs over common parts costs.

6) Capital Assets. According to the present status of the program, 100% of new machinery and building are dedicated to common parts.

8) Learning. Ratio D/A = 11.4/8 . Special attention has been given, to avoid or to reduce to a minimum double source items for manufacturing work-sharing, in order to obtain maximum learning benefits during the initial production phase.

The true significant convenience of project integration is evident by columns E_1, E_2, E_3 indicating comparative N.R.C. incurred by each company (or better, by each nation since national financial help or support to Agusta is to be considered) to develop the naval, civil and utility helicopter.
Commonality allows to divide by 3 incurred costs for the common parts.
Double partnership work-sharing allows to divide by 2 the N.R.C.'s. A typical formula to calculate the cost incidence on each variant is :

$$N.R.C.\ V_i = (C/3 + C_i)\ 1/2$$

where :

N.R.C. V_i = N.R.C. or "i" variant, National share
C = common parts N.R.C. (column C)
C_i = add-one N.R.C. peculiar to "*" variant (column C_1, C_2, C_3).

At the bottom of E_i columns an amount is shown, indicating, as percentage of "A" columns, the N.R.C. at nation level to develop each variant.
As an example 40.5 (bottom of E_1 column) means that, due to simultaneous integrated development of 3 variants and to WHL-Agusta work-sharing, each nation will get a fully developed naval version of the EH 101 at 40.5% of the cost that would have been incurred developing a dedicated naval helicopter (of EH 101 class) without international partnership.

Figure 37 compares N.R.C. at national level, to be afforded in order to develop three different helicopters (naval, civil and utility) with three different approaches :

1st approach. A single company is tasked to develop three dedicated aircraft of about the same weight, but with no commonality.

2nd approach. A single company is tasked to develop three specific variants of an integrated project.

3rd approach. Same as second approach but with one foreign partner.

4. CONCLUSIONS

The EH 101 is a high performance all weather, low pilot workload multirole helicopter designed to operate in a wider range of environments and scenarios than present generation rotorcraft.
The tri-variant modular concept, with a high degree of commonallity (61% to 81% in terms of unit cost), together with the bi-national Industrial work sharing approach has allowed a saving relative to the development of three separate variants and a further splitting of non recurring cost between the two nations, while at the same time expanding the total market.
In quantitative terms the following parameters are significant :

- Total N.R.C. of a 3 variant national program would be 60% of the total cost of three monovariant programs

- Total N.R.C. of a 3 variant bi-national program is slightly higher (115%) than the corresponding national program, but since it is split into two, each nation's contribution is actually 57.5%

In order to satisfy the requirements of all roles the resulting A/C is a high performance machine often exceeding the minimum required in each single role, since it is generally driven by the most demanding of the three.
It was possible to avoid paying for these exceptional capabilities in terms of empty weight (thus reduced payload) through maximum utilization of high technology.
The overall result is surprisingly favorable : EH 101 empty weight is between 7.5% (naval) and 14% (civil) lower than the average trend.
Of course the extra non recurring costs of the introduction of advanced technology into the program are easily affordable through the above seen bi-national approach and the wide market which the EH 101 has bought for himself.

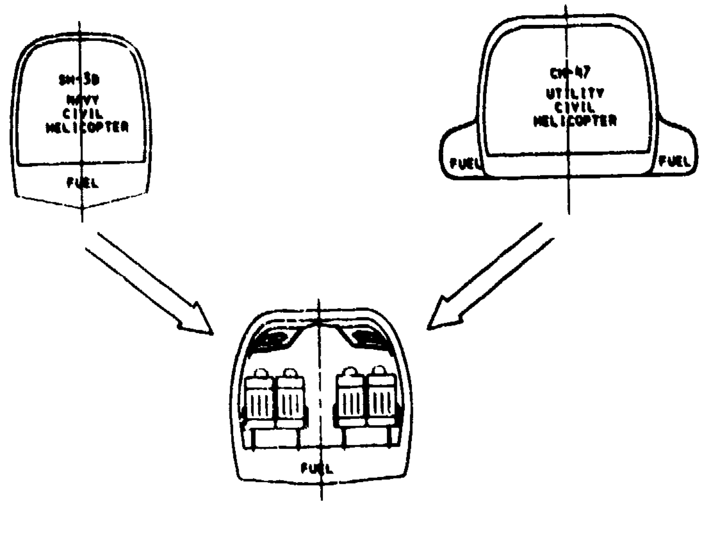

FIGURE 1 - EH101 INTEGRATED NAVAL, CIVIL & UTILITY

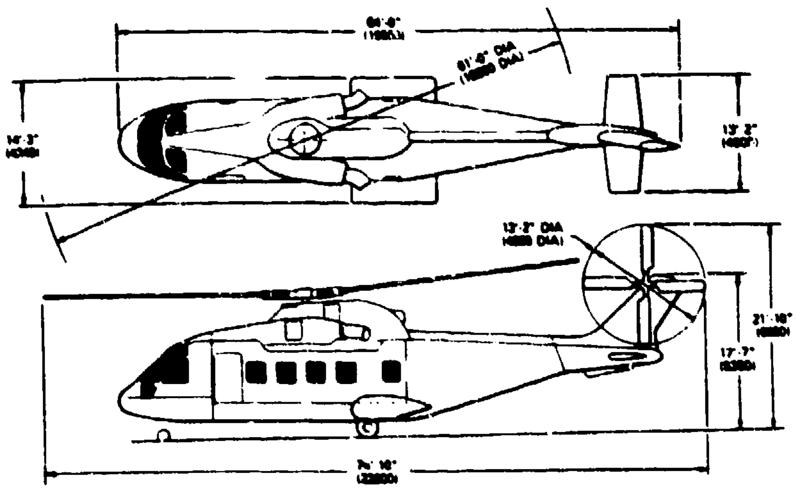

FIGURE 2 - EH101 LEADING DIMENSIONS

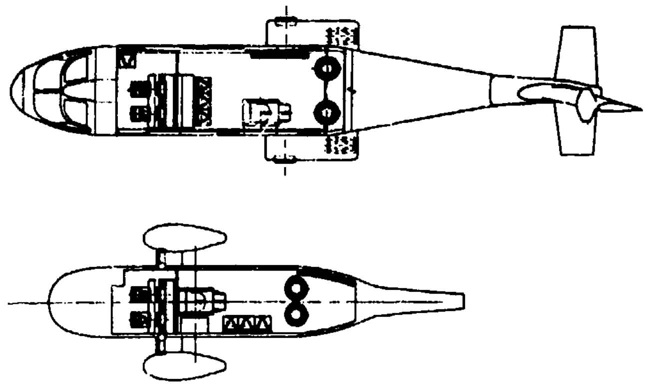

FIGURE 3 - EH101 & SH-3D WITH SAME EQUIPMENTS

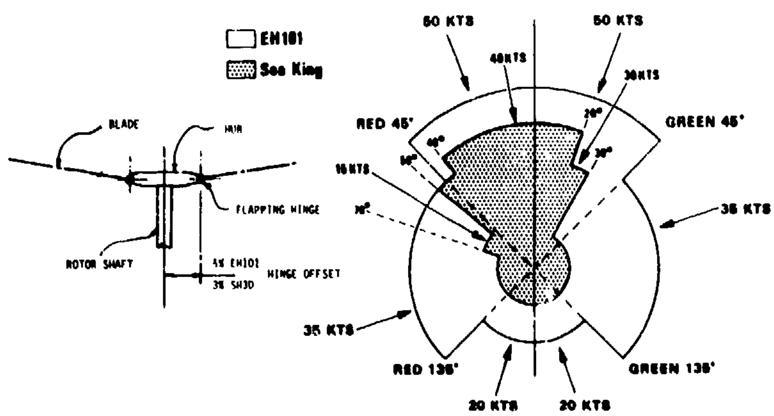

FIGURE 4 - LOW SPEED FLIGHT ENVELOPE (AGILITY)

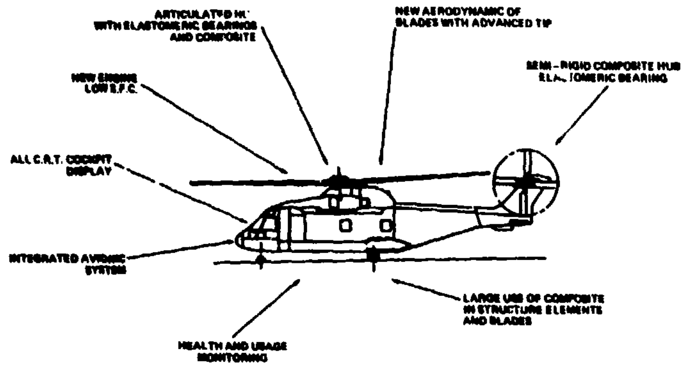

FIGURE 5 - EH101 NEW TECHNOLOGY

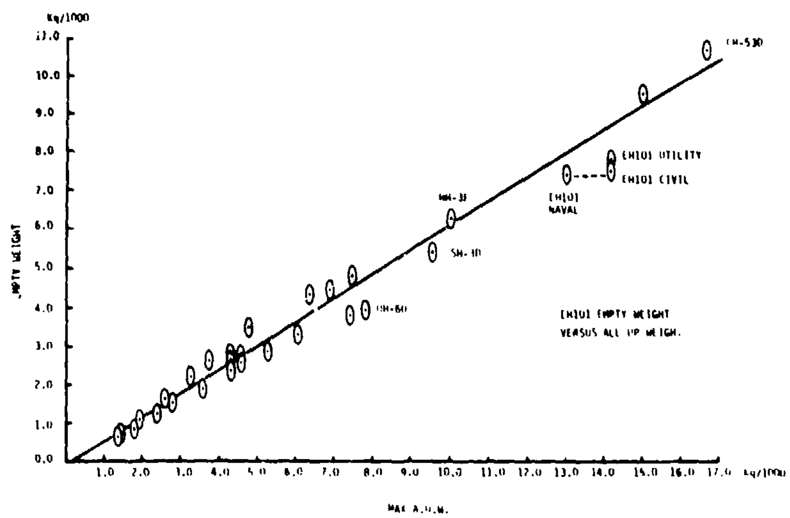

FIGURE 6 - EH101 EMPTY WEIGHT VERSUS ALL UP WEIGHT

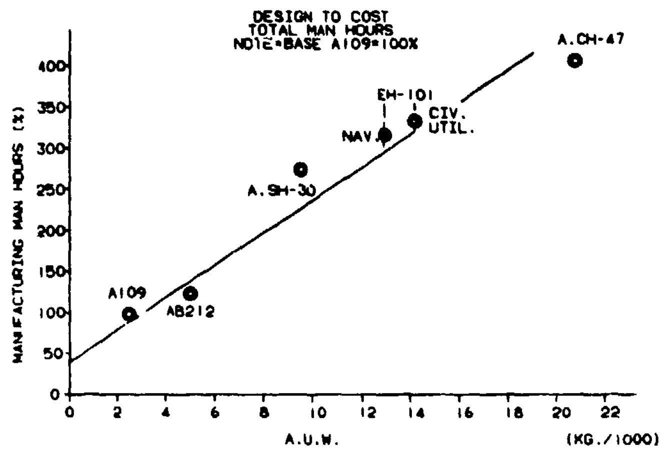

FIGURE 7 - TOTAL MAN HOURS VERSUS ALL UP WEIGHT

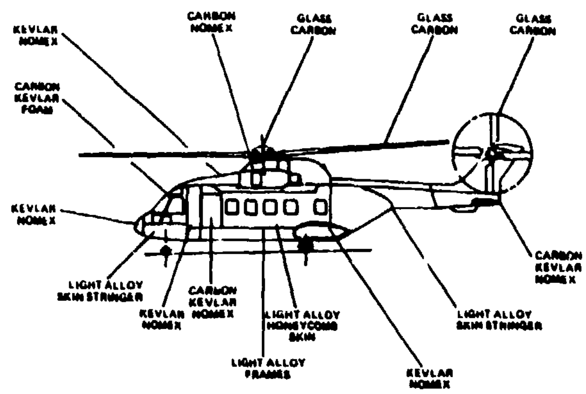

FIGURE 8 - COMPOSITES IN EH101

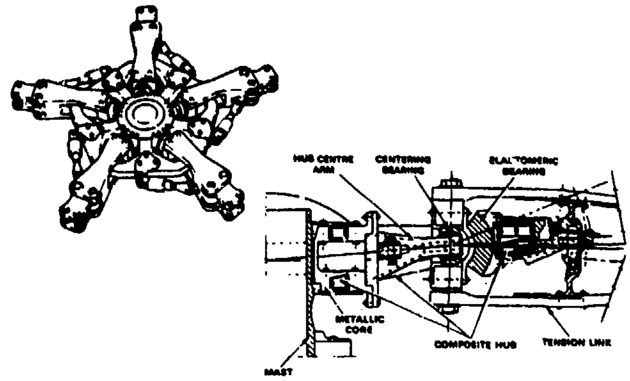

FIGURE 9 - EH101 MAIN ROTOR HUB

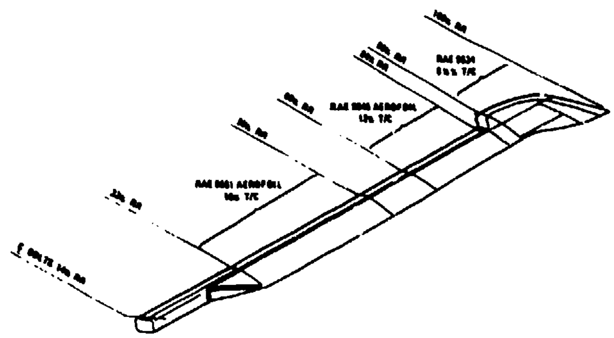

FIGURE 10 - EH101 MAIN ROTOR BLADE

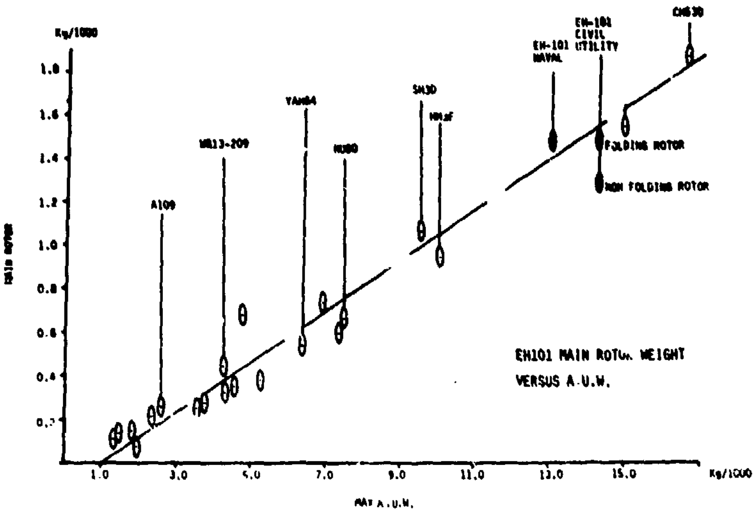

FIGURE 11 - EH101 MAIN ROTOR WEIGHT VERSUS ALL UP WEIGHT

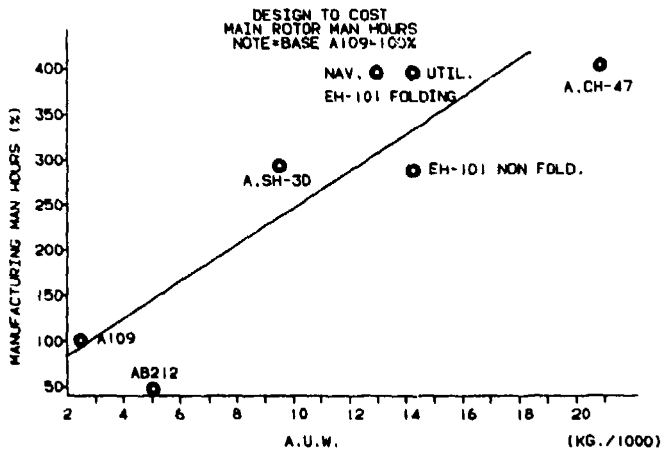

FIGURE 12 - EH101 M.R.H. MANUFACTURING HOURS VERSUS ALL UP WEIGHT

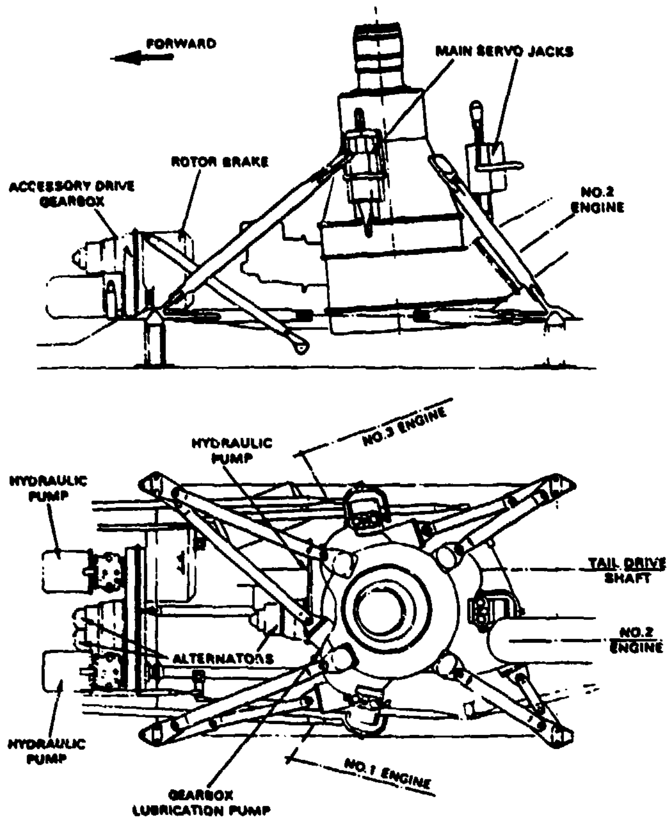

FIGURE 13 - EH101 MAIN GEAR BOX LAYOUT

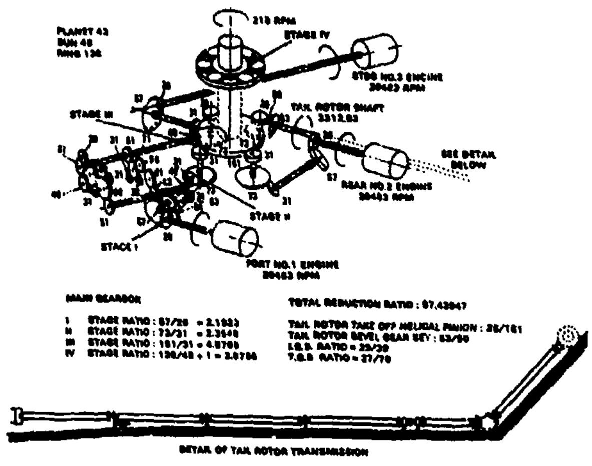

FIGURE 14 - EH101 MAIN GEAR BOX SCHEME

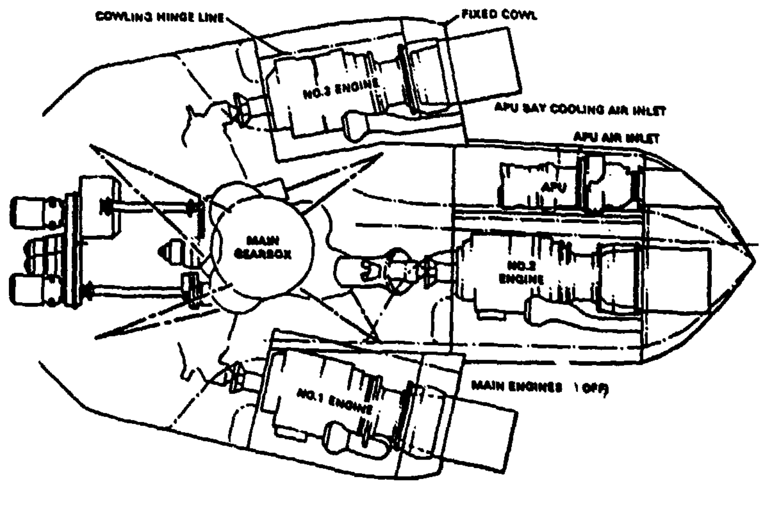

FIGURE 15 - EH101 DRIVE SYSTEM LAYOUT

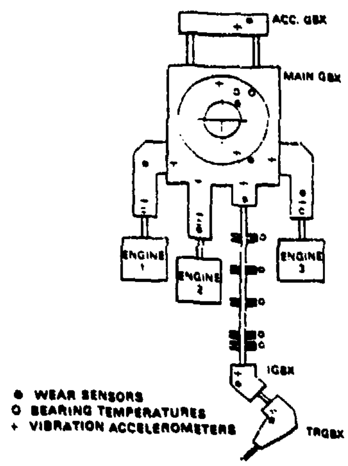

FIGURE 16 - EH101 DRIVE SYSTEM HEALTH AND USAGE MONITORING

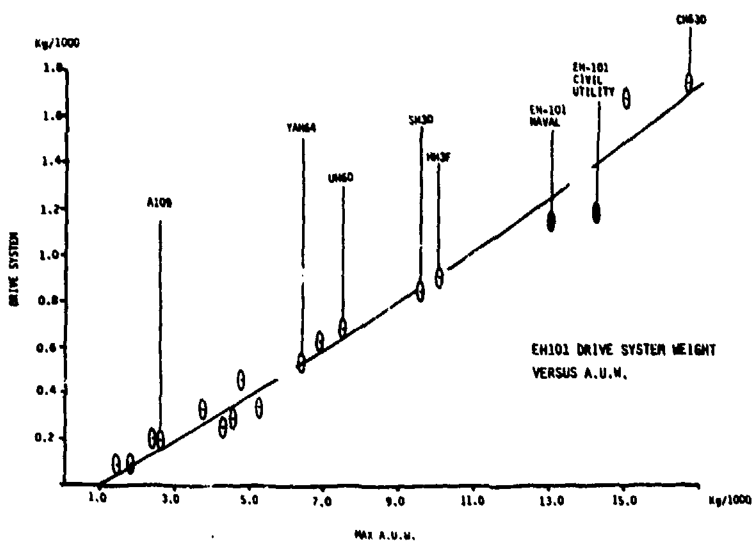

FIGURE 17 - EH101 DRIVE SYSTEM WEIGHT VERSUS ALL UP WEIGHT

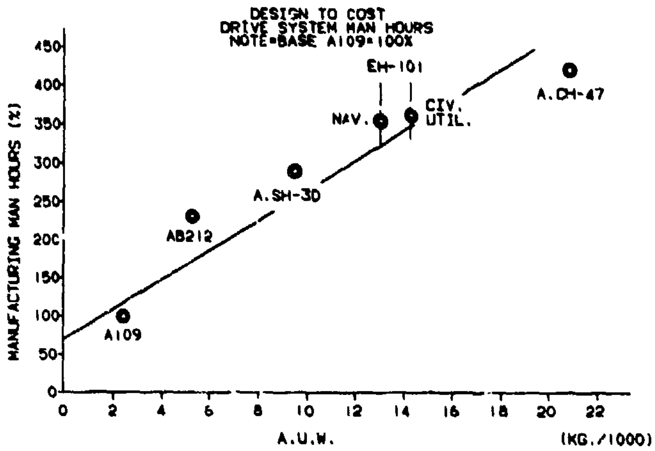

FIGURE 18 - DRIVE SYSTEM MAN HOURS VERSUS ALL UP WEIGHT

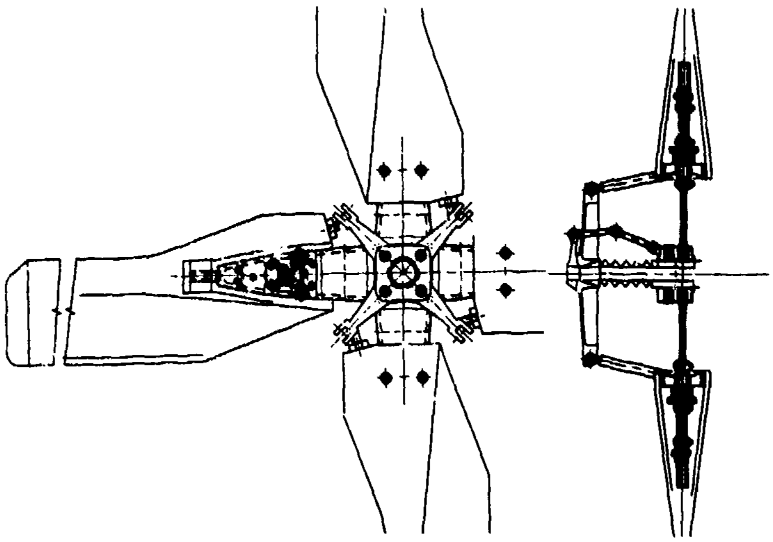

FIGURE 19 - EH101 TAIL ROTOR LAY OUT

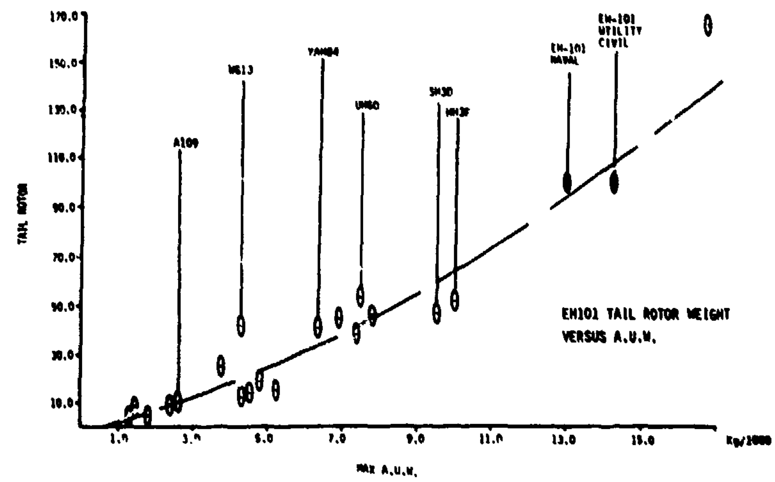

FIGURE 20 - EH101 TAIL ROTOR WEIGHT VERSUS ALL UP WEIGHT

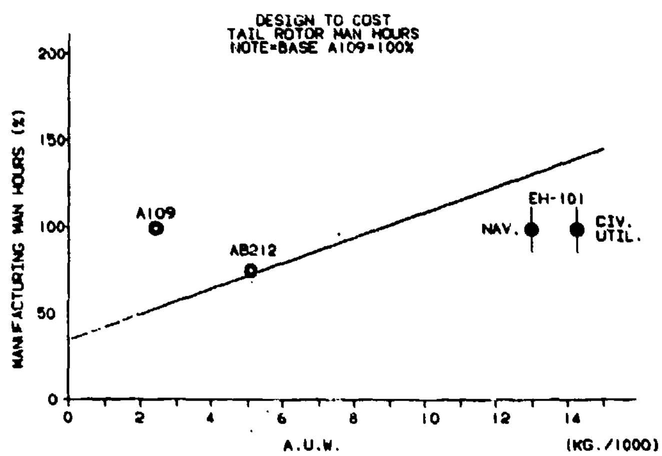

FIGURE 21 - EH101 TAIL ROTOR MAN HOURS VERSUS ALL UP WEIGHT

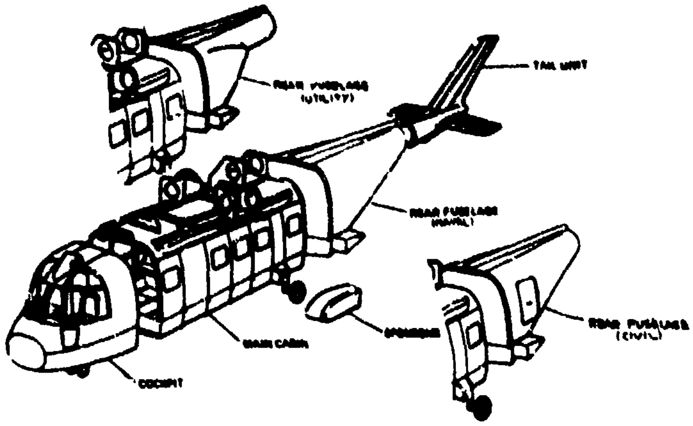

FIGURE 22 - EH101 AIRFRAME MODULAR DESIGN

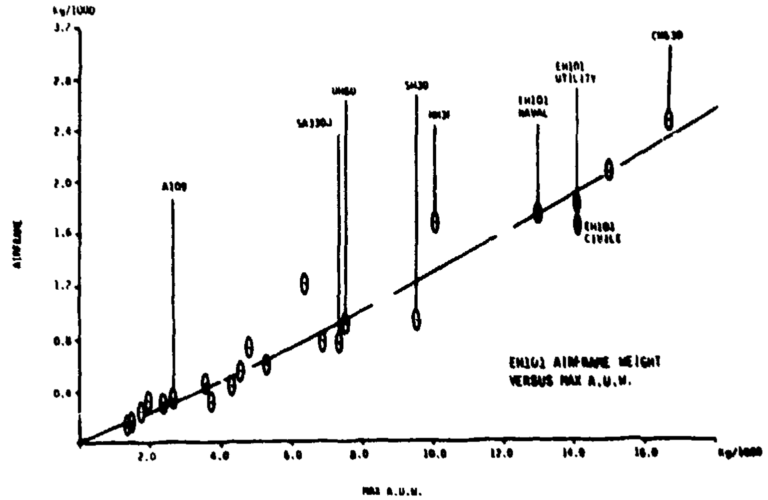

FIGURE 23 - EH101 AIRFRAME WEIGHT VERSUS ALL UP WEIGHT

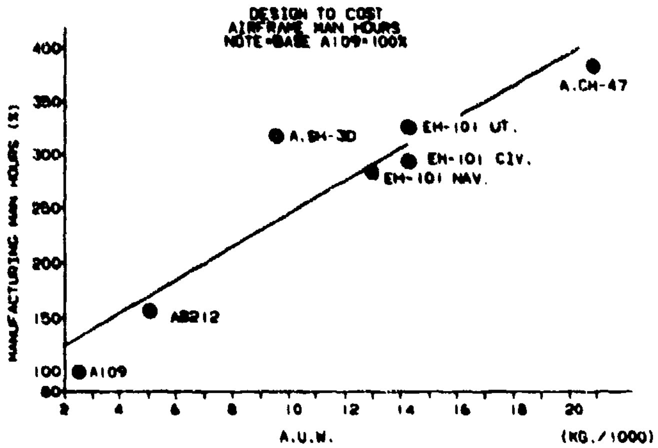

FIGURE 24 - AIRFRAME MAN HOURS VERSUS ALL UP WEIGHT

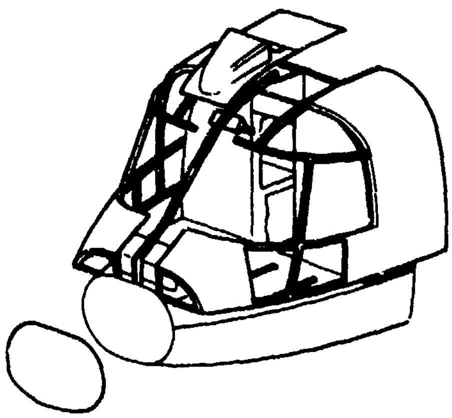

FIGURE 25 - EH101 COCKPIT DESIGN

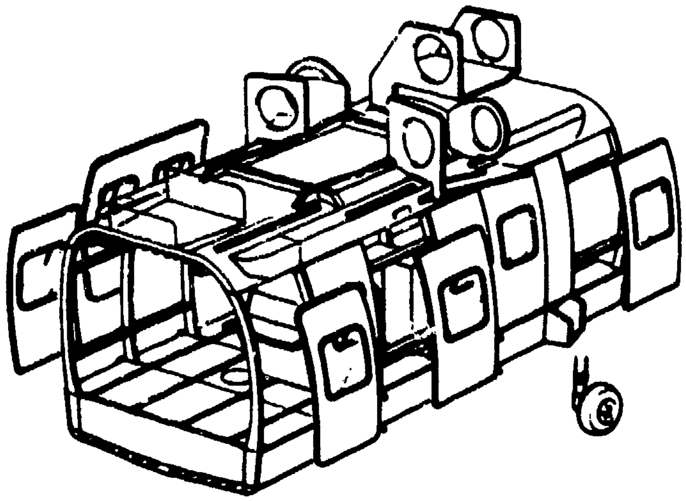

FIGURE 26 - EH101 CABIN DESIGN

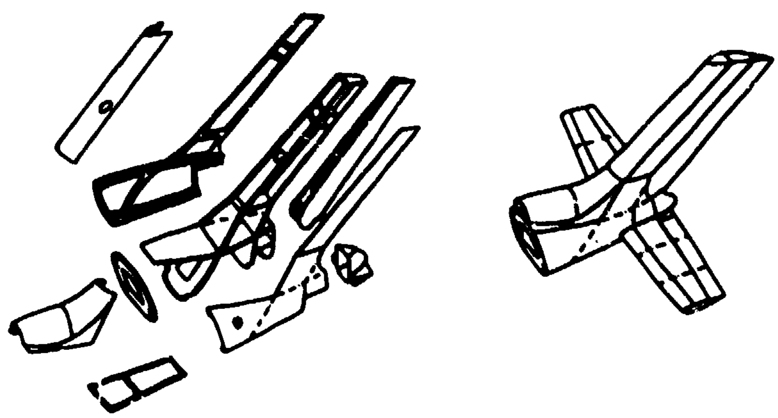

FIGURE 27 - EH101 TAIL UNIT DESIGN

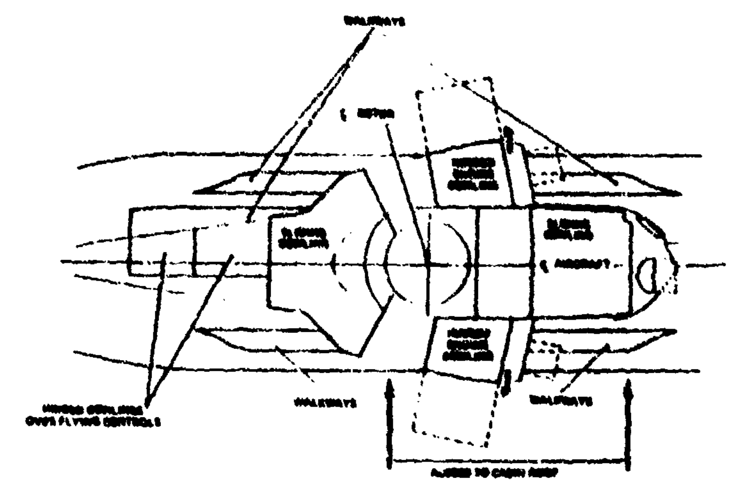

FIGURE 28 - EH101 WALKWAYS & COWLING DESIGN

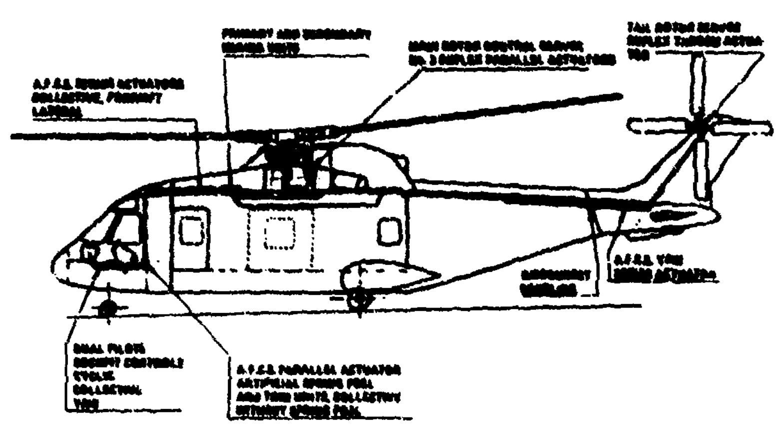

FIGURE 29 - EH101 FLIGHT CONTROLS SCHEME

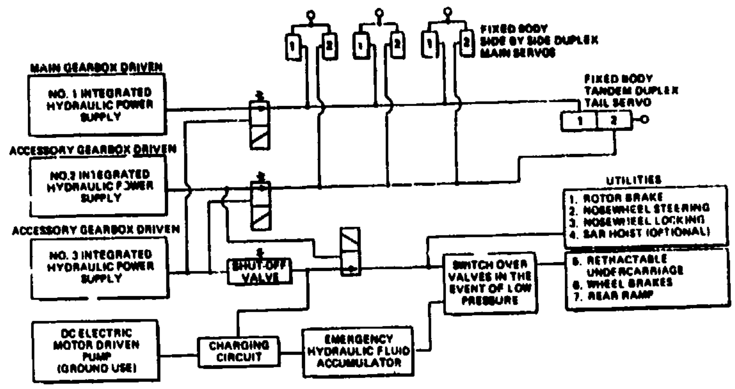

FIGURE 30 - EH101 HYDRAULIC SYSTEM SCHEME

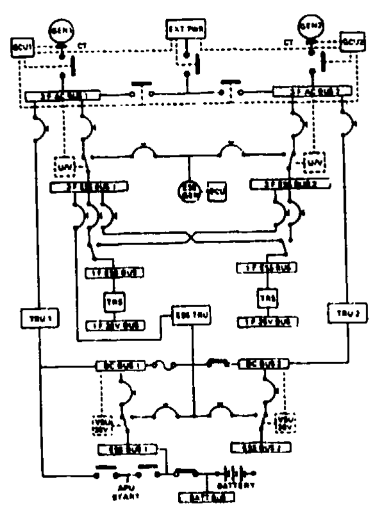

FIGURE 31 - EH101 ELECTRICAL SYSTEM SCHEME

25 30

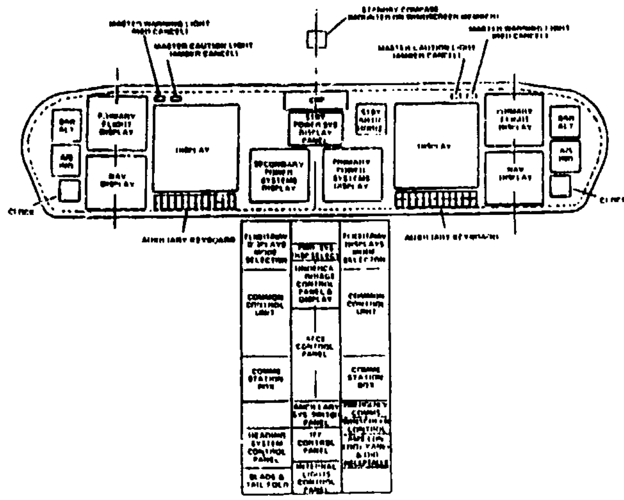

FIGURE 32 - EH101 INSTRUMENT PANEL MILITARY STANDARD

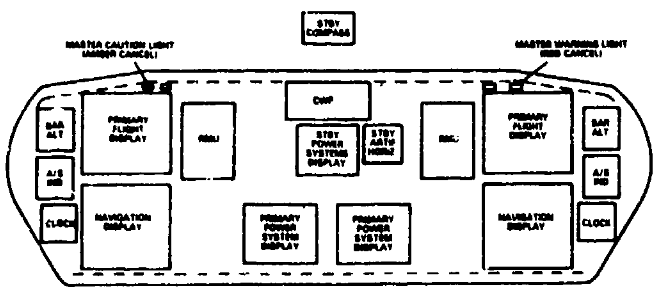

FIGURE 33 - EH1O1 INSTRUMENT PA EL CIVIL STANDARD

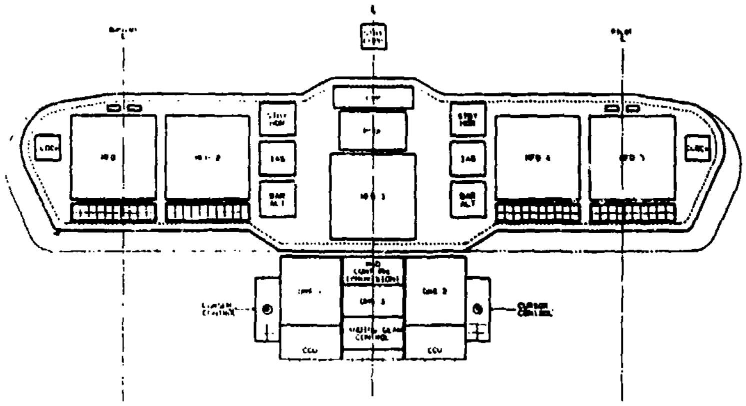

FIGURE 34 - EH101 INSTRUMENT PANEL 5 DISPLAYS OPTION (NAVAL/CIVIL/UTILITY)

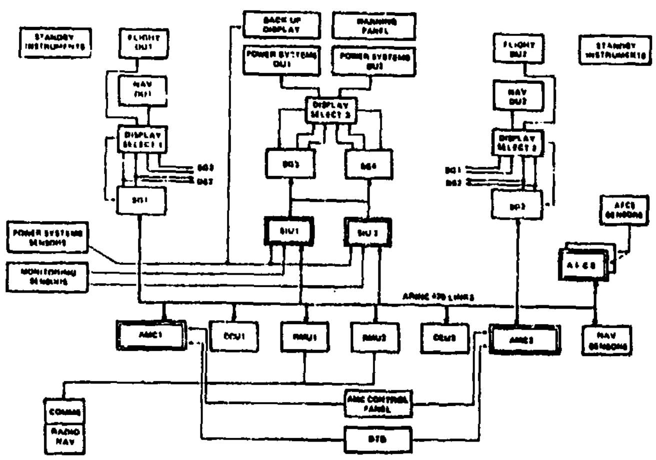

FIGURE 35 - EH101 CIVIL STD. AV¡ONIC ARCHITECTURE ARINC 429

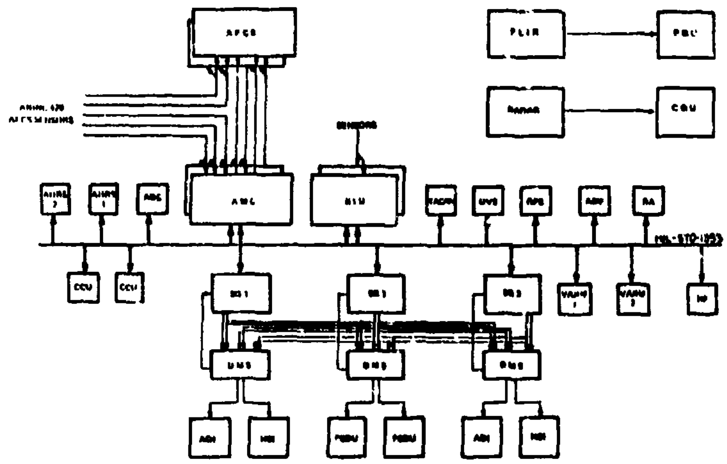

FIGURE 36 - EH101 MILITARY STD. AVIONIC ARCHITECTURE MIL-STD-1553

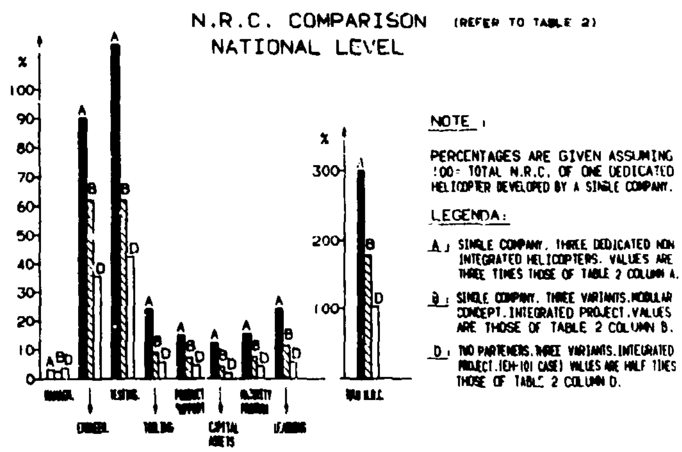

FIGURE 37 - EH101 DIFFERENT APPROACHES TO DEVELOP A NAVAL + A CIVIL + AN UTILITY HELICOPTER.

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14. Abstract

The expanding roles of the helicopter and the intensified threat perceived by its potential user have led to proposals for future rotorcraft with characteristics significantly different to existing types. The resulting rapid evolution of rotorcraft configurations, in response to user demands, now requires a translation into design criteria to permit the aerospace R&D community to provide appropriate and cost effective responses to these demands. The objective of this symposium was to explore the impact of operational needs on the evolution of rotorcraft design. The result will be to provide a review of the present status of rotorcraft design and to identify priorities and neglected topics. Three specific issues were central:

— The translation of operational mission requirements into design criteria

— The evaluation of techniques to incorporate user defined needs into the design and methods of test and verification

— The identification of design areas where unusual or new user needs are demanding special or radical features. Keywords: Symposia; Hovering; Rotary wing aircraft, Military aircraft; Aeronautical [illegible] Mission profiles; Tilt rotor aircraft; Helicopter engines; Vibration; Night flight. (edc)

All papers were obtained by invitation.

This conference and these proceedings were assessed in a Technical Evaluation Report, commissioned by the AGARD Flight Mechanics Panel and published separately as an Executive Summary as AGARD AR-243.

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